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DESIGN OF MULTI-MISSION CHEMICAL PROPULSION MODULES FOR PLANETARY ORBITERS

VOLUME III: APPENDIXES

15 AUGUST 1975

Prepared for
NASA AMES RESEARCH CENTER

under
Contract NAS2-8370

TRW
SYSTEMS GROUP

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The final report of this study is presented in three volumes:

I Summary Report

II Technical Report

III Appendixes

Use of Metric and English Units in this Report

The results of this study are reported in metric and English units. The metric notation generally is quoted first. However, since in the present transition phase most of the engineering work is still being performed in terms of English units, some of the supporting calculations are reported only in these units. In other instances English units are stated first, with metric units in parentheses, e.g., in reference to a 12-foot (3.66 meter) antenna dish.

APPENDIXES

APPENDIX A

STATE OF PROPULSION TECHNOLOGY

1. BACKGROUND

Propulsion systems to be used in the multi-mission propulsion module must satisfy criteria that are unique to the missions considered in this study, including the following:

- Mission life may approach 10 years
- Fluorine may be required as oxidizer to provide the high performance essential to the missions (high specific impulse)
- Multiple restarts are required with long dormant periods, e.g., major ΔV impulse at earth departure is followed by the planetary orbit insertion maneuver many years later
- The system must be compatible with different thermal conditions in extremely hot (Mercury orbiter) or cold (outer-planet orbiter) mission environments
- The system must conform with strict safety requirements of the Shuttle orbiter as launch platform, i.e., safety of propellant handling and storage; remote leak detection; rapid disposal of propellants by overboard dumping, etc.
- Multi-purpose use of propellants is desired, with main thrust and auxiliary thrust engines to be supplied by a common tankage and pressurization system.

A prudent design approach must be taken which satisfies the long mission life requirement without demanding extraordinary advances in technology. It must minimize risk due to possible component unreliability by adding component redundancy and functional redundancy and by avoiding sources of wearout failure.

A system with a 10-year lifetime cannot be tested practically in real time. Accelerated life tests may be performed in some instances at elevated operating temperatures, higher than normal pressure, increased cycle rates or other intensified conditions that tend to expose design weaknesses, improper materials selection, or faulty fabrication techniques. However, such a test approach may not be truly representative of failure mechanisms and combined degradation effects that occur under prolonged use in actual missions. Therefore, it is obvious that the problem of developing systems for extremely long life missions without prohibitive demonstration cost will continue to be a technical challenge.

For systems using earth-storable propellants, a primary objective is extension of the demonstrated capability from about 2 years up to about a decade. Propulsion systems using earth-storable bipropellants (N_2O_4/MMH) have demonstrated lifetimes on the order of 2 years in actual flight programs. Monopropellant hydrazine (N_2H_4) propulsion systems have a somewhat longer demonstrated life.

For space-storable systems with fluorine oxidizers the technology base is quite limited and a considerably greater advancement in the state of the art is necessary. Although technology efforts and advanced developments have been started, no fluorine system has flown thus far.

An important question relates to long-term storage and isolation of the fluorine oxidizer. A properly passivated elemental (non-alloy) metallic tank containing pure fluorine should be capable of indefinite storage. Practical considerations include effects of alloy materials in the tank, impurities in the tank and imperfections of manufacture.

Planetary orbit missions with total impulse requirements in the 3000 to 4500 m/sec class, such as the missions considered here, may well be the first missions to justify flight application of fluorine propulsion. Other applications may then follow.

The applicable technology, including materials, components, engine characteristics, cooling techniques, and feed systems will be reviewed here. Areas where additional development is required will be indicated.

2. TECHNOLOGY STATUS

Table A-1 summarizes the technology status, or state of the art, that existed as of 1974 and forms the basis of this study.

For earth-storable systems, the state of the art is represented by systems using cold-gas pressurized N_2O_4 and MMH with pressure-fed

Table A-1. Initially Assumed Specific Impulse and Propulsion Module Inert Weight Data

Item	Propellant Type	
	N_2O_4/MMH	F_2/N_2H_4
Specific impulse (For ϵ = 52, F = 600 lb_m, 2730 N)	296 sec	363 sec (demonstrated) 376 sec (anticipated)
Propulsion module inert weight	$W_i = 0.163\ W_p + 27.2$ kg	$W_i = 0.163\ W_p + 36.3$ kg
(for total mass between 1000 and 6000 lb_m; 441 and 2720 kg)	Does not include mission-peculiar equipment such as sun shades, etc. (see later revision, Section 7)	
Mass fraction	≈0.82	≈0.82

Note: ϵ = nozzle expansion ratio
F = thrust force

ablative, conduction or radiation cooled engines operating at 100 to 200 psi (7 to 14 bar) chamber pressures. Spacecraft propulsion systems utilizing this propulsion technology include TRW's Multi-Mission Bipropellant Propulsion System (MMBPS); Mariner and Viking propulsion systems of the Jet Propulsion Laboratory (JPL); NASA's Apollo Service Module, Lunar Descent (LMDE) and Lunar ascent propulsion systems; the Titan Transtage and several reaction control systems (RCS). The MMBPS, Mariner, and Viking are those most similar to the systems considered in this study.

No space-storable propulsion systems have been flown or even qualified. Much of the recent interest in such systems has been at the Jet Propulsion Laboratory, where in-house and sponsored work aimed at planetary retropropulsion applications has been conducted for several years.

JPL has sucessfully tested a complete (although not flight-weight) fluorine propulsion system at their facilities at Edwards Air Force Base, California, with good success (Reference 1).

The technology baselines used in this study are defined as follows:

1) Earth-storable (N_2O_4/MMH)

   a) TRW MMBPS

   b) JPL Mariner (Reference 16)

2) Space-storable (LF_2/N_2H_4)

   a) JPL F_2/N_2H_4 test propulsion system (Reference 1)

   b) TRW design or in NAS7-750 (Reference 4)

   c) F_2/N_2H_4 engines as described in "Comparison Study of Fluorine/Hydrazine Engine Concepts" performed for JPL NAS7-100 PO 953943 (Reference 17)

   d) F_2/N_2H_4 engine experience at TRW

   e) F_2 compatibility as described in "Compatibility Testing of Spacecraft Materials and Space-Storable Liquid Propellants" performed for JPL by TRW under NAS7-100 task order RD-31 and 93 (Reference 18)

   f) Other liquid fluorine experience as described in the literature.

Table A-2 summarizes applicability of the data base. Pertinent characteristics of the baseline technology for N_2O_4/MMH and LF_2/N_2O_4, are summarized in Tables A-3 and A-4, respectively.

3. ENGINE TECHNOLOGY

3.1 Typical Characteristics

The state of the art in N_2O_4/MMH engines includes both radiation-cooled and regeneratively-cooled engines in the size and chamber pressure range of 100 to 1000 lb_f (445 to 4450 N) and 80 to 200 psi (5.5 to 13.8 bar). Radiation-cooled engines, in general, are lighter than ablative engines. Five examples of existing engines are given in Table A-5.

Characteristics of a rocket engine under development by Marquardt for the Space Shuttle RCS application are also shown in Table A-5. Its film-cooled columbium combustion chamber operates at a throat temperature of 1800 to 2200°F (980 to 1200° C) and is designed for very long life. Operating parameters are optimized for the Space Shuttle mission and are not typical of an engine designed for a planetary orbiter mission.

Radiation-cooled columbium chambers have been successfully used in vacuum with throat temperatures of at least 2500°F (1370°C) at a chamber pressure of around 100 psia (7 bar). One engine with a molybdenum chamber is quoted as operating at 100 lb_f (445 N) thrust at 170 psia (11.7 bar) chamber pressure, with specific impulse of 290 seconds and a throat temperature of 2500°F (1370°C). Operating temperature of up to approximately 2500°F (1370°C) is thus considered the state of the art of 1974 for radiation-cooled N_2O_4/MMH engines.

3.2 Cooling Techniques

Combustion chamber cooling techniques on engines in the range of interest are embodied primarily by two types of chambers: radiation-cooled or silica-phenolic ablatively-cooled chambers, with or without throat inserts. The lighter radiation cooling approach is preferred if suitable for the configuration. In both cases boundary-layer film cooling is used as a supplementary cooling method but with a resulting

Table A-2. Spacecraft Propulsion System Data Base Used in Study.

	Earth-Storable Systems (N_2O_4/MMH)	Space-Storable Systems (F_2/N_2H_4)
Propulsion system	MMBPS* (TRW) Mariner Mars '71 (JPL) Apollo lunar module descent stage (TRW) Published literature	No F_2/N_2H_4 flight systems Published literature TRW space-storable thermal control technology study (under JPL contract) TRW propellant isolation shutoff valve study (under JPL contract)
Flight experience	Extensive (TRW, JPL, and others)	Oxidizer F_2: negligible Other cryogenic: extensive with LO_2 Fuel N_2H_4: extensive Other amine bipropellants: extensive
Engines	TRW family of scalable engines: lunar module descent engine, MMBPS Mariner '71	TRW advanced developments JPL advanced developments
Materials	Established technology	TRW F_2 materials compatibility test program (under JPL contract)
Ground operations	TRW flight programs Mariner	TRW test site experience TRW and JPL monopropellant flight programs (fuel side, not LF_2) Published literature

*MMBPS - Multimission bipropellant propulsion system (TRW)

Table A-3. Characteristics of State-of-the-Art N_2O_4/MMH Propulsion System Technology

Component Area	Mariner Mars '71	Other Available Technology
1. Propellant containment material	Heat treated 6Al-4V titanium σ_y = 160 - 175,000; SF_B = 2	Aluminum; cryoformed stainless steel
2. Pressurant containment	Annealed 6Al-4V titanium 4000 psig, σ_u = 135,000 psi; σ_y = 125,000 psi; SF = 2	
3. Pressurant isolation	Pyrotechnic actuated shears parent metal	
4. Propellant isolation	Pyrotechnic actuated shears parent metal	
5. Propellant acquisition	Bladders and standpipe	Centrifugal action; surface tension
6. Engine operating modes	Bipropellant	Bipropellant/monopropellant dual mode (bimodal)
7. Engine cooling method	Boundary layer/conduction — radiation nozzle; I_{sp} = 288 sec	Radiation cooled, ablative; regenerative; I_{sp} = 296 sec
8. Thermal control	Absorptivity/emissivity control	Electric heaters; radioisotope heating units
9. Micrometeoroid protection		Metal honeycomb; quartz fabric
10. Structure	Beryllium tube truss; magnesium and steel fittings	Titanium truss; aluminum fittings

Note: σ_y = yield strength
σ_u = strength ultimate
SF_B = burst safety factor

Table A-4. Technology Applicable to (or in Advanced Development for) Space-Storable (F_2/N_2H_4) Propulsion Systems*

Propulsion System Component Area	Assumed as Baseline	Other Technology
1. Propellant containment	CRES stainless steel, 6Al-4V titanium - alloy	6Al-4V titanium alloy 2219 aluminum or nickel liner
2. Propellant isolation	Aluminum and gold metal-to-metal seals	
3. Propellant acquisition	Active expulsion devices not applicable to LF_2 tank; use settling rocket (non-spinner) or centrifugal action (spinner)	
4. Engine operating modes	Bipropellant (liquid-liquid)	Dual mode (gas-liquid combustor) possible
5. Engine cooling method	Ablative with throat insert	Radiation-cooled graphite with barrier cooling
6. Propellant thermal control	Thermal shielding for LF_2 tanks (insulation alone not sufficient)	
7. Thermal control	Absorptivity/emissivity control by zones on tank	
8. Micrometeoroid protection	Silica fabric cover (Beta cloth)	Metal honeycomb or foils
9. Insulation	Closed-cell PBI foam on LF_2 tanks, multilayer insulation on N_2H_4 tanks	

*Entries apply to LF_2 (oxidizer) part of system, exceptions noted

Table A-5. Characteristics of Existing Earth-Storable Bipropellant Engines

	MMBPS	Shuttle RCS	MBB Symphonie	Mariner 71	P-50 I_{SPS}
Propellant	N_2O_4/MMH	N_2O_4/MMH	N_2O_4/A 50	N_2O_4/MMH	HDA/USO*
Thrust, N (lb_f)	391 (88)	2880 (872)	391 (88)	1317 (296)	396 (89)
Specific Impulse (sec)	295	290	303	287	272
Chamber Pressure Bar (psi)	6.2 (91)	10.3 (152)	7 (102)	8 (115)	6.4 (94)
Nozzel Area Ratio	52:1	22:1	77:1	40:1	52:1
Weight kg (lb_m)	4.54 (10)	6.6 (14.5)	1.95 (4.3)	7.8 (17.1)	3.5 (7.7)

* HDA (High Density Acid) - 54% HNO_3/44% N_2O_4
USO (Lockheed designation) - 99% UDMH/1% silicon oil

loss in specific impulse performance. These engines have used earth-storable propellants (N_2O_4/MMH or similar). Radiation-cooled engines have been limited to about 100-psia chamber pressure. Cooling of LF_2/N_2H_4 is accomplished predominantly with carbon or graphite liners, often with addition of silica-phenolic backup layers.

3.3 LF_2/N_2H_4 Engines

Considerable experience with the LF_2/N_2H_4 propellant combination has been accumulated. However, this cannot compare with the experience gained on the many flight systems which use earth-storable propellants. A considerable amount of testing with LF_2/N_2H_4 was conducted in the 1950's and 1960's. Recent tests have used heat-sink and carbon-containing liners such as pyrolytic graphite or carbon fibers (e.g., Carb-i-tex combinations). These are more durable under exposure to the reaction products of LF_2/N_2H_4 than are silica materials.

3.4 Dual Mode Engines

Dual mode, also called bimodal, engines are also considered in this study. A dual mode engine operates either on bipropellants or alternatively, on N_2H_4 monopropellant. Flexibility achieved by bimodal operation offers such advantages as: 1) small impulse maneuvers can be accomplished accurately, 2) propellants can be settled without acquisition devices in the oxidizer tank, and 3) in the case of systems using N_2H_4 as fuel, reserve propellant can be tanked and used either for velocity or attitude maneuvers without advance apportionment to either mode of engine operation. For the propulsion systems considered in this study, the conventional bipropellant (or liquid-liquid) engine is adopted. Principal reasons for this selection are design conservatism and uncertainty regarding prospects of dual-mode engine development.

3.5 Auxiliary Thruster State-of-the-Art

Several auxiliary thrusters are presently available in the size range of interest (see Table A-6). Monopropellant hydrazine is the state-of-the-art propellant for low thrust engines, although a flight system using N_2O_4/Aerozine-50 has been developed in Europe.

Table A-6. Examples of Candidate Auxiliary Thrusters

	Monopropellant Thrusters	Bipropellant Thrusters	
		Symphonie (European)	Technology Program
Propellant	N_2H_4	N_2O_4/A-50	N_2O_4/MMH
Status	Qualified	Qualified	Under Development
Thrust levels (lb_f)	0.35 to 1.2	2 to 3	2 to 5
(N)	(1.6 to 5.5)	(9.1 to 13.6)	(9.1 to 22.7)
Minimum impulse bit (lb_f-sec)	0.03	0.04	0.04
(N-sec)	(0.14)	(0.18)	(0.18)
Specific impulse (sec)	212 to 230	293	290 to 300
at steady state (typical) (sec)	220		290
with minimum impulse bit (sec)	110		200 to 220

Performance of the auxiliary thrusters in the pulsed mode is a function of pulse duration as illustrated in Figure A-1 and A-2 for monopropellant and bipropellant, respectively. Selection of appropriate auxiliary thrusters for the mission in question depends on the state of development and on system performance tradeoffs.

In the thrust class of less than 5 lb_f (22.2 N) and for near-term applications, with a development cycle of only a few years, the flight proven N_2H_4 monopropellant thrusters are the best choice. A N_2O_4/MMH bipropellant system with a 2 to 5 lb_f thrust level (8.9 to 22.2 N) has undergone a considerable amount of testing and may become operational within a few years. A similar European-developed 2.2 lb_f (10 N) bipropellant thruster using N_2O_4/Aerozine 50 is being used on the German-French Symphonie satellite.

In the class of less than 1 lb_f (4.5 N) of thrust, monopropellant N_2H_4 thrusters are the best choice, considering their low cost and high reliability, even at the low I_{sp} level (200 to 220 sec) characteristic of these thrusters. In the propulsion module using F_2/N_2H_4, the hydrazine can serve as monopropellant for the auxiliary thrusters. For the earth-storable (N_2O_4/MMH) systems, auxiliary thrusters of the bipropellant type can be used if a 2 to 5 lb_f (8.9 to 22.2 N) thrust level is acceptable.

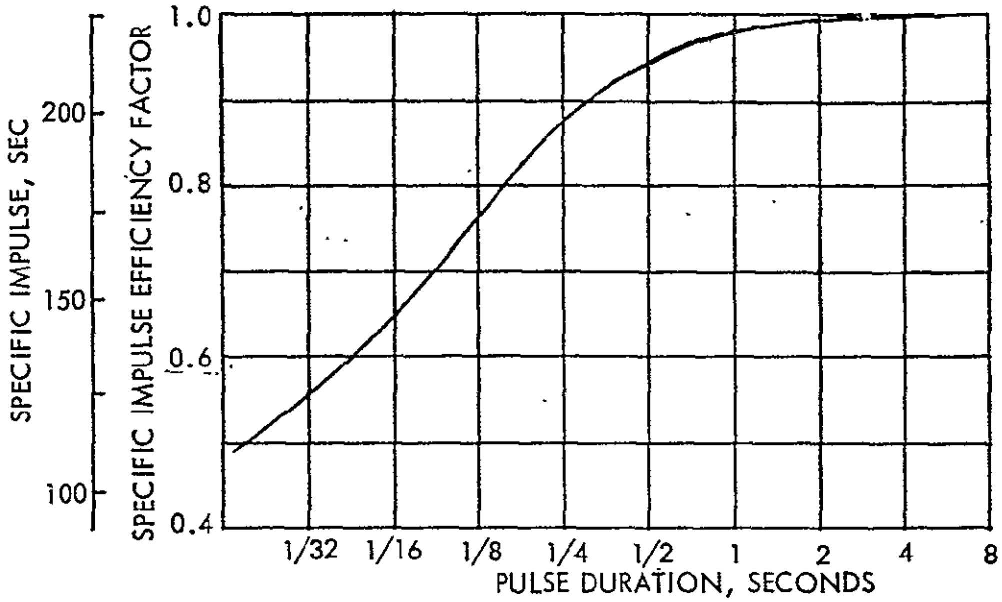


Figure A-1. Monopropellant Thruster Specific Impulse Efficiency Versus Pulse Duration

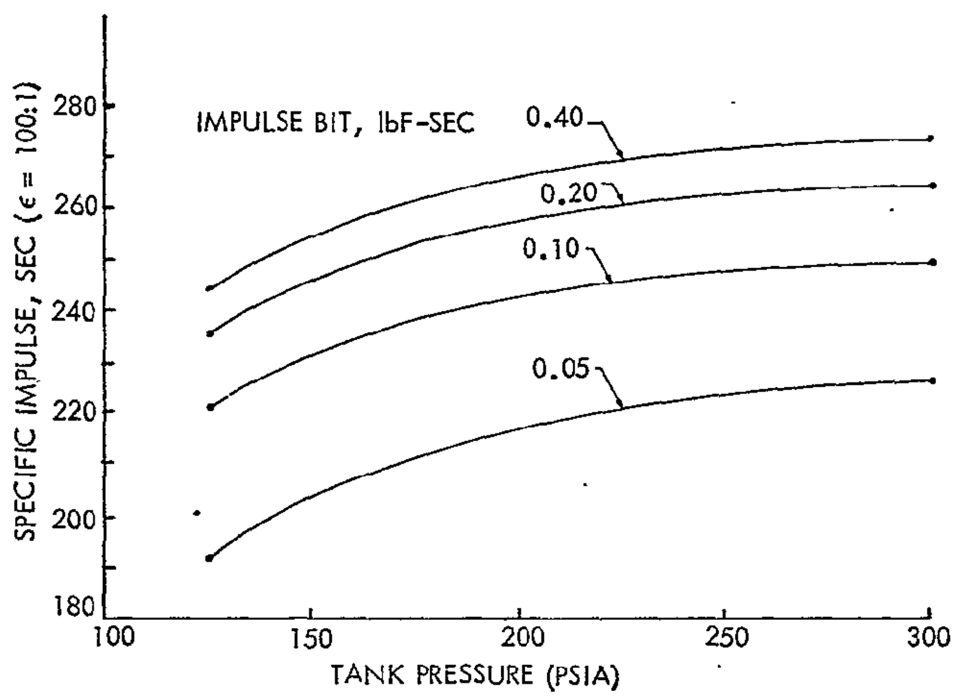


Figure A-2. Bipropellant Thruster Pulse Mode Performance — Typical

APPENDIX B

SUMMARY OF FLUORINE SYSTEM SAFETY CONSIDERATIONS AND LAUNCH SITE ASSEMBLY SEQUENCE

This appendix presents a summary of results of related studies dealing with Shuttle safety implications with regard to loading, transporting and launching of payloads that include liquid fluorine as propellant, and with launch site assembly procedures. This material augments the discussion of LF_2 handling and storage presented in Section 5 of Volume II.

1. SAFETY IMPLICATIONS

Results of a concurrent study of Shuttle safety implications performed by TRW (Reference 8) are directly applicable to this study and were used in assessing safety characteristics and providing safety features of the space-storable propulsion system. The following paragraphs give a brief summary of the objectives of that study and the results obtained.

The study objectives were:

1) To identify any unique system requirements and constraints imposed by the use of LF_2 as oxidizer in the propulsion system of a planetary spacecraft launched by Shuttle orbiter.

2) To compare the safety interfaces between the Shuttle (crew and hardware) and the spacecraft propulsion system when LF_2, instead of N_2O_4, is used as oxidizer.

The primary hazard to personnel is leakage of LF_2 during propellant loading operations. Loading is similar to routine transfers of LF_2 from tanker trucks to industrial user facilities. The operations involved should be isolated from locations where personnel and facilities are concentrated.

Transportation and installation of the loaded propulsion module rank next in the list of potential hazards. Safety of these operations can be improved by applying stricter regulations and standards than those currently adhered to when transporting the chemical on public highways. Regarding the installation of the loaded propulsion module onboard the

Shuttle cargo bay, regular safety requirements must be enforced and careless handling (e.g., high shock loads) is ruled out.

If the propulsion system has been loaded, transported, and installed in accordance with strict safety requirements and procedures, and if external hazards from other systems in the Shuttle cargo bay are minimized, any residual hazards during normal flight operations appear low. Clearly, the risk of performing a Shuttle abort and emergency landing with a large quantity of liquid fluorine onboard would be too high and dumping provisions must be made available to dispose of the fluorine along with the other propellants (e.g., those carried by the Shuttle upper stage) that also must be dumped prior to an abort. To handle the dumping procedure of LF_2 during Shuttle orbital operations is comparable to dumping of other hypergolic propellants except for requiring a specially treated (passivated) dump line.

The overall rationale for accepting the risks inherent in using LF_2 as oxidizer in Shuttle-launched interpalanetary spacecraft compared with N_2O_4 is summarized as follows:

1.) The likelihood of accidents involving N_2O_4 is comparable to and at least not higher than when this oxidizer is carried for other uses, particularly for the Shuttle orbit maneuvering system (OMS Kits), because in the spacecraft propulsion module there are fewer and smaller tanks, and no external lines containing the oxidizer.

2) The likelihood of accidents involving LF_2 can be made comparable to N_2O_4 or even lower through stricter safety provisions.

3) In both cases the chance of accidents can be made remote by adhering to strict safety standards in all phases of handling and operation.

Key safety recommendations of the referenced study are summarized as follows:

- Isolate oxidizers by confinement in tanks only, i.e., eliminate oxidizers from pipes while in transit
- Use all-welded construction and double-walls for propellant tanks
- Provide appropriate remote propellant loading facilities

- Automate leak detection and warning at the launch site
- Institute appropriate safeguards and handling procedures at the launch site and during flight
- Provide appropriate safety features on the Shuttle orbiter, especially to prevent hazards from other systems
- Provide liquid nitrogen cooling of the LF_2 tanks until liftoff
- Provide propellant status instrumentation and display to the Shuttle crew
- Provide a dump system for immediate safe disposal of all propulsion module propellants in the event of leak or other unsafe conditions; also, if integrity of the LF_2 or N_2O_4 tanks is threatened by malfunction of other systems; and in preparation for a mission abort.

A second study recently completed by TRW Systems under contract with NASA, Kennedy Space Center (Reference 31) covered the various phases of ground processing of Shuttle payloads that use fluorine propulsion stages. The study confirmed the feasibility of processing such systems for launch by the Shuttle orbiter without undue safety hazards and without significant impact on the environment (ecology). The study defines ground processing and ground safety criteria that must be adhered to when handling the toxic, corrosive and highly flammable chemical, and compares these requirements with the conventional safety provisions that apply in handling nitrogen tetroxide (N_2O_4). It recommends development of caution-and-warning sensors to be installed at the assembly and loading stations and onboard the Shuttle orbiter and the further development of protective clothing for ground support personnel.

2. PROPELLANT LOADING

Loading of propellant presumably occurs at a location remote from the Space Shuttle launch pad 39. The loading operation consists of:

- Receiving the propulsion system from the point of manufacture and inspecting it for damage
- Ensuring that the fluorine components are "fluorine-clean"
- Passivating the system with first diluted and then pure fluorine gas

- Chilling down the tank with LN_2 in the cooling coil
- Loading LF_2 by gravity feed or by cryopumping as the tank is chilled by LN_2
- The loaded propulsion system is capped and transported to a storage shed pending installation into the Shuttle orbiter.

Storage should be at a temperature near the LF_2 normal boiling point of -306°F. The normal boiling point of LN_2 is -321°F, which allows a convenient margin.

3. LAUNCH SITE ASSEMBLY SEQUENCE

The selected baseline sequence corresponds to Option 3 identified in previous JPL and TRW studies (References 8 and 32). This option, even though the most difficult to implement, was selected because it is the safest. The specific sequence is as follows (also see Figures B-1 and B-2):

1) Either the interim upper stage or Tug (IUS/Tug), or whatever upper stage is used, is installed horizontally in the orbiter cargo bay. This is done in the Orbiter Processing Facility (OPF). The orbiter will then be erected in the Vehicle Assembly Building (VAB) and transported in a vertical position to launch pad 39A or B. Also, the upper stage has an interstage truss installed in the OPF.

2) The Payload Changeout Facility (PCF) is used to install solid propellant kick stages to eliminate safety hazards to the OPF or VAB. This may affect the timeline as the PCF will not be available to accommodate the spacecraft and its propulsion until after the solid rocket is installed. The kick stage is attached to a thrust case which is mounted to the interstage truss.

3) When the Shuttle upper stages are ready, the Pioneer or Mariner type spacecraft and integrated propulsion module(s) will be transported to the pad, disconnected from their coolant supply in the case of LF_2, and hoisted into the PCF. Cooling will then be reconnected.

4) The flight spacecraft will be installed within the cargo bay, and cooling reconnected through the lines which enter the cargo bay via the umbilical.

5) The spacecraft will be joined at all disconnect points and through its field joint (interface) to the IUS/Tug. (Resume LN_2 cooling and check out the GHe prechill cooling mode.)

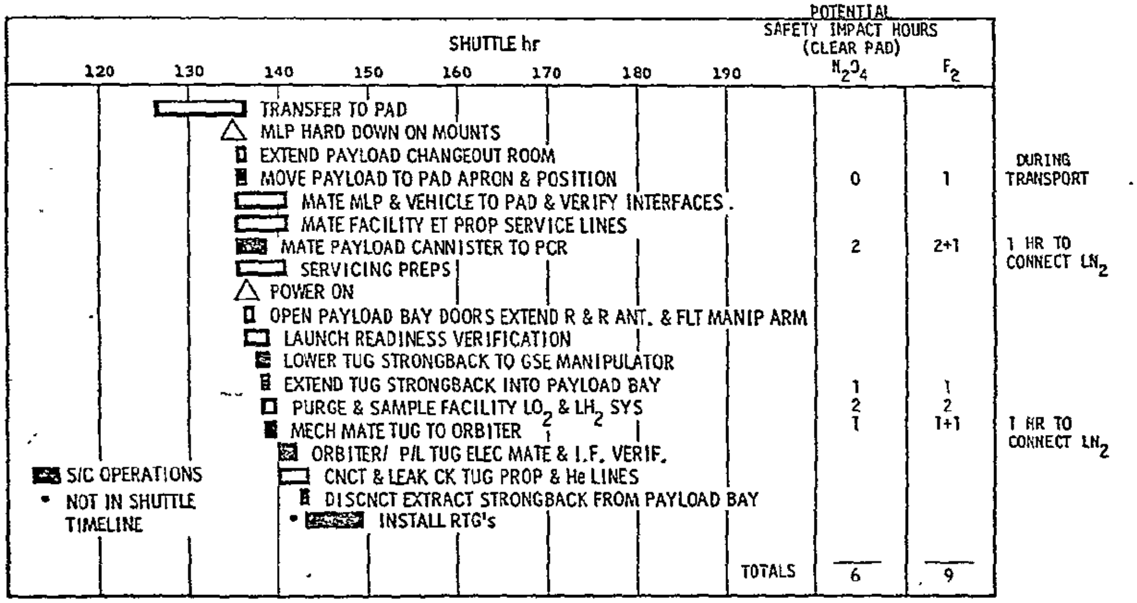
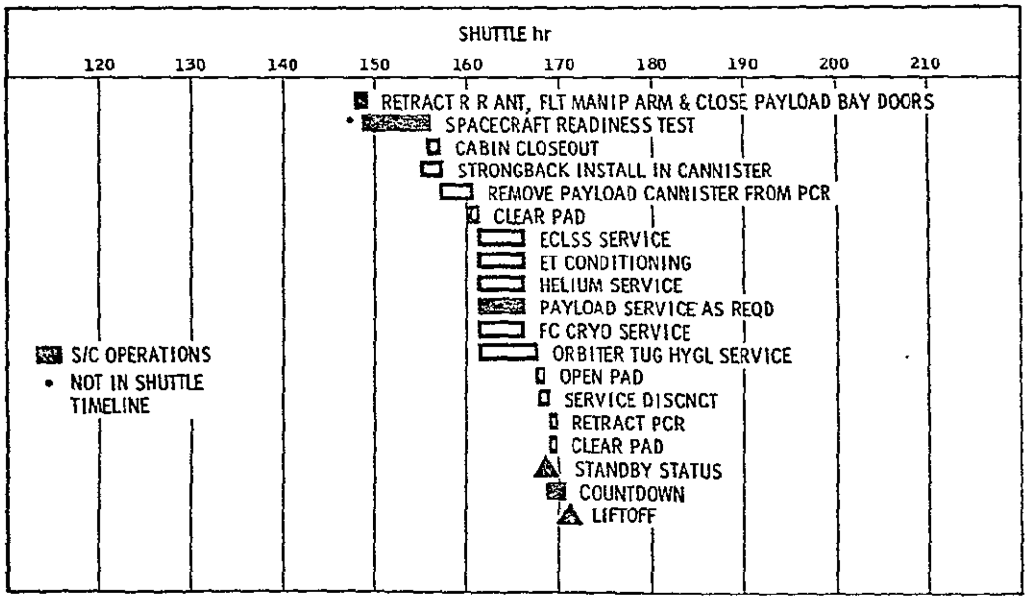


Figure B-1. Launch Pad Operations - Payload Installation on Pad, RTG Installation on Pad



LEGEND

MLP – MOBILE LAUNCH PLATFORM (CRAWLER)

RR ANT. – RENDEZVOUS RADAR ANTENNA

GSE – GROUND SUPPORT EQUIPMENT

I.F. VERIF. – INTERFACE VERIFICATION

ECLSS SERVICE – ENV. CONTROL & LIFE SUPPORT SYSTEM SERVICE

ET CONDITIONING – EXTERNAL TANK CONDITIONING

FC CRYO SERVICE – FUEL CELL CRYOGENIC SYSTEM SERVICE

HYGL SERVICE – HYPERGOLIC PROPULSION SYSTEM SERVICE

Figure B-2. Launch Pad Operations - Payload Installation on Pad, RTG Installation on Pad (continued)

6) When cooling of the fluorine tanks has resumed, checkout of the spacecraft to IUS/Tug interface will be performed.

7) Flights using fluorine and carrying Dump Kit Peculiar Fluorine (DKPF) lines from the spacecraft interface through the orbiter are passivated with gaseous fluorine.

8) The Shuttle cargo bay doors are closed.

9) Other operations preparatory to launch are accomplished as shown in Figure B-1 (Reference 5, JPL Study) including cabin closeout, orbiter external tank propellant servicing (loading) and IUS/Tug hypergolic propellant servicing (loading), etc., prior to launch.

10) After doors are closed and prior to the scheduled launch the LF_2 cooling may be changed from normal LN_2 to GHe pre-chill mode to provide greater heat soak capability in the propellant.

A possible variation to the above is currently being investigated to provide more convenient access to separation joints on the flight spacecraft and interstage adapter. This involves steps 1 through 4 of the sequence. Instead of mating the flight spacecraft to the interstage adapter (and possibly solid propellant motor) already installed on the IUS/Tug in the Shuttle orbiter bay, these units are mated first outside the bay, and then installed in the bay together.

APPENDIX C

STRUCTURAL ANALYSIS

This appendix presents the structural analysis documentation including stress analyses and preliminary weight assessments for four configurations:

1) Tandem Pioneer 1. This configuration includes a 750-pound Pioneer spacecraft supported by a pair of tandem propulsion modules that use earth-storable propellants for an inbound mission.

2) Tandem Pioneer 2. Same configuration as Pioneer 1 except that it is sized for the lower-volume space-storable propellant.

3) Tandem Mariner 1. This configuration includes a 1210-pound Mariner Spacecraft supported by a pair of tandem propulsion modules that use earth-storable propellants for an inbound mission. Also included is an adapter between the spacecraft and upper module.

4) Tandem Mariner 2. Same configuration as Mariner 1 except that it is sized for the space-storable propellant.

TANDEM PIONEER (1)

- 750-lb_m PIONEER S/C
- EARTH STORABLE PROPELLANT
- 800-lb_f ENGINE

①

TRW
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PREPARED JAGliksman 4/21/75 REPORT NO. PAGE 1

CHECKED

MODEL MMCPMPO

Pioneer Payload ①

Tandem Module
Earth Storable Propellant

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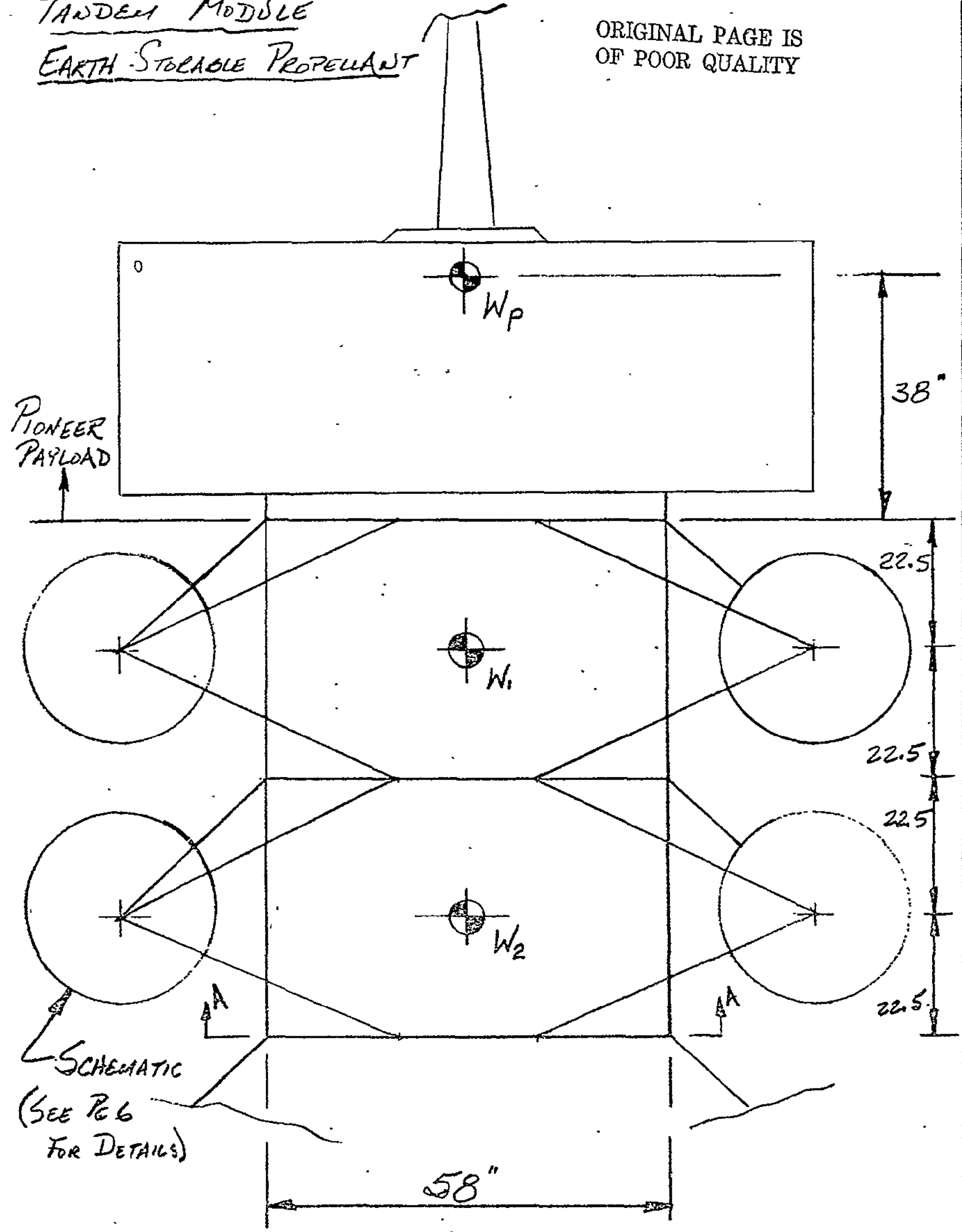


SYSTEMS 1441 REV. 8-67

TANDEM MODULE - (EARTH STORABLE)

ASSUMING THAT THE MODULES ARE IDENTICAL, $W_1 = W_2$ & STRUCTURE IS THE SAME, THEN ONLY THE LOWER MODULE NEEDS ANALYSIS.

W_p = PAYLOAD WEIGHT = 750 lbs.

$W_1 = W_2$ = MODULE WEIGHT* = 3120 lbs.

*EACH MODULE CONTAINS 2550 lbs OF PROPELLANT $\therefore W_1^* = W_2^* = 570$ lbs EMPTY

CRITICAL LOADING CONDITIONS - (ULTIMATE)

CONFIGURATION	CONDITION	g_{AXIAL}	$g_{LATERAL}$
EMPTY	① CRASH Δ	9.00	0
EMPTY	② CRASH Δ	0	4.50
EMPTY	③ LANDING	±1.20	4.26
FULL	④ BOOST	-4.95	1.17
FULL	⑤ LIFT-OFF	-4.35	2.70

Δ NOTE: CONDITIONS ① & ② ARE NOW (4/16/75) ASSUMED TO ACT SIMULTANEOUSLY.

TRW
SYSTEMS GROUP
ONE SPACE PARK • REDONDO BEACH, CALIFORNIA
PREPARED JA GLIKSMAN 4/21/75 REPORT NO.
CHECKED
MODEL MMC PMPO

PIONEER PAYLOAD ①

LOADS @ SECTION A-A - (ULTIMATE)

$P_{AX} = g_{AX} \times \Sigma W_{TS.}$

$M = \underline{128} \times g_L \times W_P + 67.5 \times g_L \times W_1 + 22.5 \times g_L \times W_2$

CONDITION	P_{AX}	M	P_{EQ}^*
①†	17,010	0	17,010 Δ
②†	0	662,900	45,720 Δ
③†	± 2270	621,500	±45,550
④	- 34,600	440,900	- 65,000
⑤	- 30,400	1,017,400	- 100,600

$^*P_{EQ} = P_{AX} + \frac{2M}{R}$ where $R = 29$ IN.

∴ THE CRITICAL LOADING CONDITION FOR SHELL BUCKLING IS CONDITION 5.

$P_{EQ} = 100,600$ lbs (ULT)

† USES EMPTY WEIGHTS

Δ ASSUMPTION OF SIMULTANEOUS LOADING (REF PG 2) WOULD ONLY PRODUCE A $P_{EQ.}$ OF 62730 lbs WHICH IS LESS THAN CONDITION 5.

SHELL BUCKLING - ALUMINUM

$R = 29$ in $\quad E = 10 \times 10^6$ psi

TYP $t = .090$ in

$R/t = 322$

ANALYSIS REF SEIDE, MORGAN et al PG 39
(TRW REPORT EM 10-26)

$C = .24$

$P_{cr} = 2\pi C E t^2$

$= 2\pi(.24)(10 \times 10^6)(.090)^2$

$= 122{,}100$ lbs

(BUCKLING) $M.S. = \frac{122100}{100{,}600} - 1 = \underline{.21}$

SEPARATION SYSTEM

MAXIMUM TENSION LOAD ALONG SEPARATION
JOINT = W lb/in

$$W = \frac{P}{2\pi R} + \frac{M}{\pi R^2} \times (.80)^*$$

$$= .0055P + .000303M \quad \text{for } R = 29 \text{ in.}$$

* FACTOR TO ALLOW FOR MODULUS EFFECT
IN MC/I DISTRIBUTION

SEPARATION SYSTEM (CONT)

$W_H = W \tan 20°$ (ASSUMES $\mu = 0$ & 20° RAMP ANGLE)

CONDITIONS 1 & 2 COMBINED ARE CRITICAL (SEE NOTE PG 2)

$W = 295$ lb/in
$W_H = 170$ lb/in. } (ULT)

REQUIRED BAND LOAD, P_B

$P_B = 2 W_H R$
$= 2(170)(29) = 9860$ lbs

THIS IS THE LIMIT BAND LOAD (PRELOAD LEVEL).

$P_{B\,(ULT)} = 1.5\,P_B = 14790$ lbs. (ULT)

∴ A HEAO TYPE (ALSO MSS) SEPARATION NUT SHOULD BE USED. ALSO USE THE HEAO SEPARATION BAND.

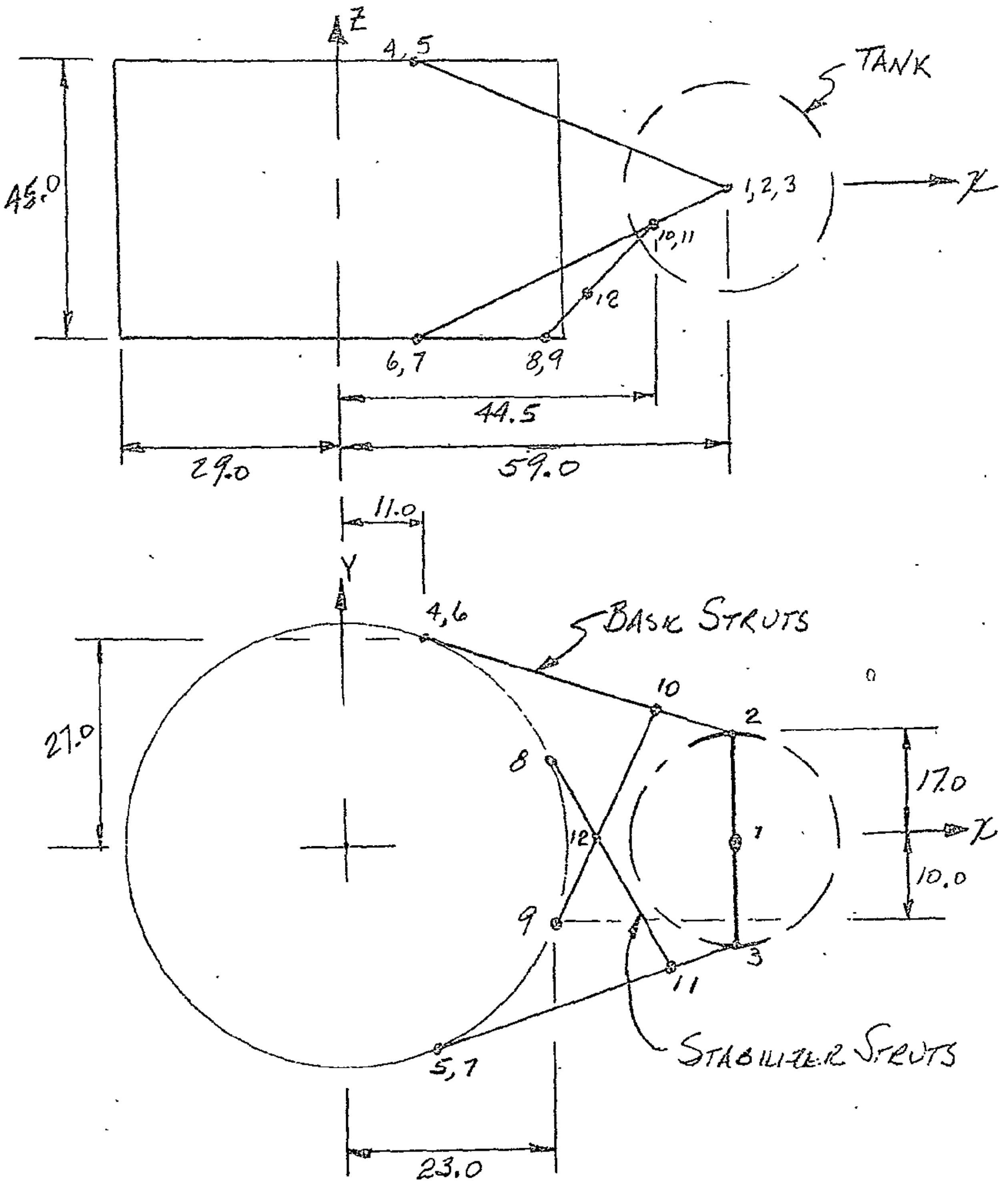
PRELOAD ALLOWABLE = 12500 lbs. } REF C117634
ULTIMATE ALLOWABLE > 20000 lbs } NUT ASSY SPEC.

(PRELOAD) $M.S. = \frac{12500}{9860} - 1 = \underline{\underline{.26}}$

(TENSION) $M.S. = \frac{20000}{14790} - 1 = \underline{\underline{.35}}$

TANK SUPPORT STRUTS

EACH TANK (LOADED) WEIGHS ≈ 700 lbs.



TANK STRUTS (CONT)

BASIC STRUTS

3.0 IN. DIA × .040 IN. 6AL-4V TITANIUM

$A = .371 \text{ IN}^2$

$I = .407 \text{ IN}^4$

CRITICAL CONDITION : NO. 5 REF TC 2

$P_{AX} = 2666$ lbs.
$M_{MAX} = 14826$ IN-lbs (ULT) (MEMBER 7-11 @ END 11)

$$f_b = \frac{P}{A} + \frac{Mc}{I} = \frac{2666}{.371} + \frac{14826(1.5)}{.407}$$

$= 61830$ psi (ULT)

$F_{cy} = 120000$ psi

$F_{ccr} = P_{cr}/A$

$R/t = 1.5/.040 = 38$ $C = .40$ (REF TRW REPORT EM 10-26)

$P_{cr} = 2\pi C E t^2 = 2\pi(.40)(17\times10^6)(.040)^2$

$= 68300$ lbs

$F_{ccr} = 68300/.371 = 184000$ psi $\therefore$ USE F_{cy}

$$M.S. = \frac{120000}{61830} - 1 = \underline{.94}$$

Tank Struts (cont)

Stabilizer Struts

2.0 in Dia × .040 in 6AL-4V Titanium

$A = .246 \text{ in}^2$

$I = .118 \text{ in}^4$

Critical Condition: No. 5 Ref Pg 2

$P_{Ax} = 1768$ lbs
$M = 237$ in-lbs } (ULT) (Member 9-12 @ End 12)

$$f_b = \frac{P}{A} + \frac{Mc}{I} = \frac{1768}{.246} + \frac{237(1.0)}{.118}$$

$= 9200$ psi (ULT)

$F_{cy} = 120000$ psi

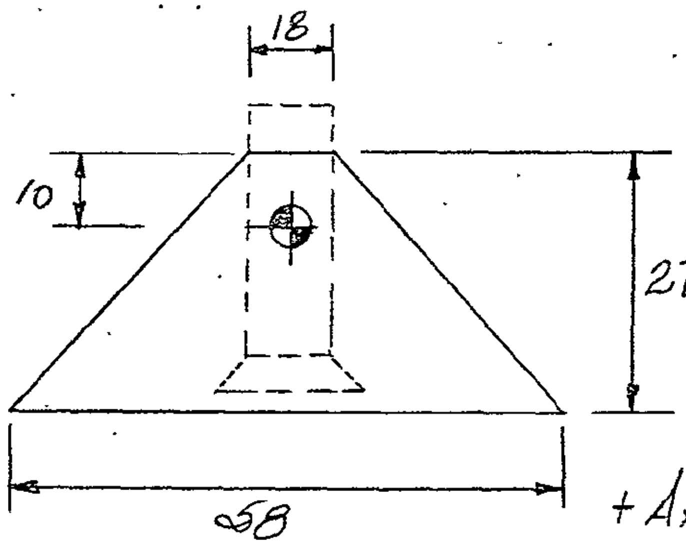
$$M.S. = \frac{120000}{9200} - 1 = \underline{\text{High}}$$

SYSTEMS 441 REV. 8-67

(9)

ENGINE SUPPORT CONE

ENGINE WEIGHT = 60 lbs.
MAXIMUM THRUST = 800 lbs. (LIMIT LOAD)



+ AXIAL LOAD IS TENSION

LOADS @ TOP OF CONE - (ULTIMATE)

CONDITION		PAX	M	PLAT
1	CRASH	540	0	0
2	CRASH	0	2700	270
3	LANDING	± 72	2556	256
4	BOOST	-291	702	70
5	LIFT-OFF	-261	1620	162
6	ENGINE THRUST	1200	0	0

ENGINE SUPPORT CONE (CONT)

USING AN ALUMINUM CONE (6061-T6)

$E = 10 \times 10^6$ psi

$t = .025$ in.

ALLOWABLE BENDING MOMENT = 32000 in-lbs
ALLOWABLE AXIAL COMPRESSION = 3500 lbs*
ALLOWABLE SHEAR LOAD = 2800 lbs
REF. TRW REPORT EM10-26

USING LOAD RATIOS:

R = APPLIED LOAD / ALLOWABLE LOAD

COND	R_{AX}	R_M	R_{LAT}	M.S. $\left(\frac{1}{\Sigma R} - 1\right)$
1	.154	0	0	HIGH
2	0	.084	.096	HIGH
3	.021	.080	.091	HIGH
4	.085	.022	.025	HIGH
5	.075	.051	.058	HIGH
6	.343	0	0	1.91

* RIVET BEARING CRITICAL

SYSTEMS '44' REV. 8-67

(1)

PREPARED JA GLIKSMAN 4/21/75 REPORT NO. PAGE 11

CHECKED

MODEL MHCPHPO

PIONEER PAYLOAD (1)

WEIGHTS - (EARTH STORABLE) - STRUCTURE

	WTS.
(1) CYLINDER - 1	
ALUMINUM t = .090 IN. L = 45 IN. D = 58 IN.	73.8
(2) SEPARATION SYSTEM - 1	
BANDS, SHOES, RETAINER, ORDNANCE (SAME AS HEHO)	35.0
(3) SEPARATION I/F RINGS - 2	
ALUMINUM X-SECT AREA = 1.00 IN^2	36.4
(4) BOTTLE SUPPORT TRUSS - 4	
A) 4 STRUTS 3.0" x .040 L = 53.9 IN (TITANIUM)	54.4
B) 2 STRUTS 2.0" x .040 L = 40.8 IN (TITANIUM)	13.7
(5) ENGINE SUPPORT CONE - 1	
ALUMINUM t = .025 IN	8.1
Σ =	221.4
+20% (UNCERTAINTY PLUS FITTINGS)	44.3
TOTAL	265.7 lbs

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TANK WEIGHTS

USING 300 psi TANKS

$R = .17$ IN

$f = \frac{pR}{2t}$

FOR $F_{TU} = 160000$ psi (6AL-4V TITANIUM)

$t = \frac{pR}{2F_{TU}}$ $p = 300 \times 2.22 = 666$ psi (BURST)

$= \frac{666(17)}{2(160000)}$

$= .036$ IN. (MIN) (USE $t = .040 \pm .004$)

SURFACE AREA = $4\pi R^2$

Weight = SURFACE × t × w

$= 4\pi(17)^2(.040)(.17)$

$= 24.7$ lbs

USE 30 lbs TO INCLUDE TEAR DROP WELD ZONE & PORTS.

GROSS WEIGHTS

TOTAL WEIGHTS - (EARTH STORABLE)

① STRUCTURE	265.7 lbs
② PROPELLANT	2550 lbs
③ TANKAGE (4x30 lbs)	120 lbs
④ PLUMBING	*
⑤ THERMAL INSULATION	*
⑥ HELIUM TANK & HELIUM 2% OF PROPELLANT (PER WHIT)	51 lbs
⑦ ENGINE	60 lbs
	3047 lbs

* NOT INCLUDED IN PRELIMINARY WEIGHT ESTIMATES

\+ CONTINGENCY

ALLOCATION = $\frac{\text{S/C GROSS} - \text{PAYLOAD}}{2}$

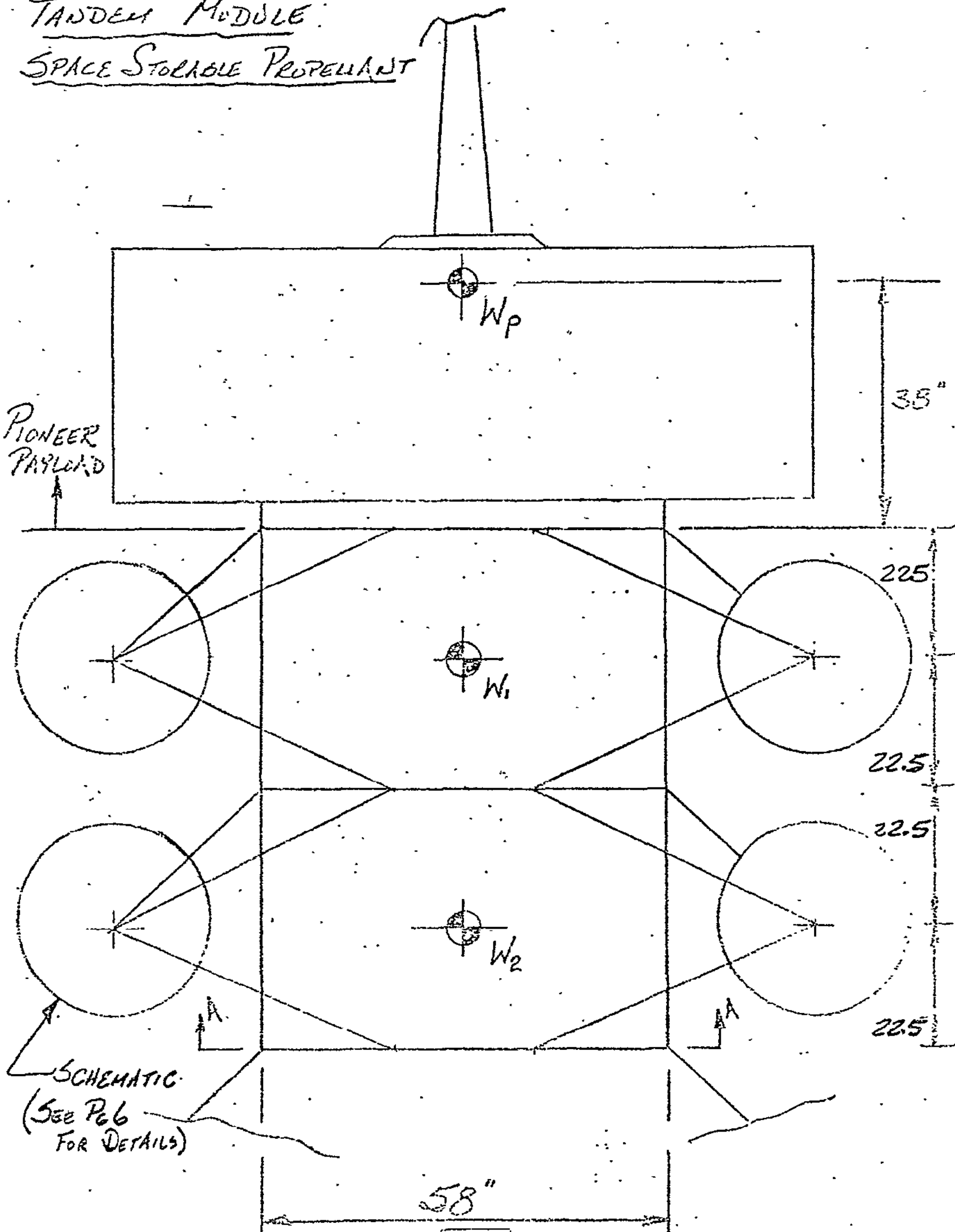
$= \frac{6990 - 750}{2} = \underline{\underline{3120 \text{ lbs}}}$

TANDEM PIONEER (2)

- 750-lb_m PIONEER S/C
- SPACE STORABLE PROPELLANT
- 800-Lb_f ENGINE

(1)

TANDEM MODULE
SPACE STORABLE PROPELLANT



TANDEM MODULE - (SPACE STORABLE)

ASSUMING THAT THE MODULES ARE IDENTICAL, $W_1 = W_2$ & STRUCTURE IS THE SAME, THEN ONLY THE LOWER MODULE NEEDS ANALYSIS.

W_P = PAYLOAD WEIGHT = 750 lbs.

$W_1 = W_2$ = MODULE WEIGHT* = 1700 lbs.

*EACH MODULE CONTAINS 1346 lbs OF PROPELLANT ∴ $W_1^* = W_2^* = 354$ lbs EMPTY

CRITICAL LOADING CONDITIONS - (ULTIMATE)

CONFIGURATION	CONDITION	g_{AXIAL}	$g_{LATERAL}$
EMPTY	① CRASH △	9.00	0
EMPTY	② CRASH △	0	4.50
EMPTY	③ LANDING	±1.20	4.26
FULL	④ BOOST	-4.95	1.17
FULL	⑤ LIFT-OFF	-4.35	2.70

△ NOTE: CONDITIONS ① & ② ARE NOW (4/16/75) ASSUMED TO ACT SIMULTANEOUSLY.

PIONEER PAYLOAD (2)

LOADS @ SECTION A-A - (ULTIMATE)

$P_{AX} = g_{AX} \times \Sigma W_{TS.}$

$M = 128 \times g_L \times W_P + 67.5 \times g_L \times W_1 + 22.5 \times g_L \times W_2$

CONDITION	P_{AX}	M	P_{EQ}^{*}
①$^+$	13120	0	13120$^\Delta$
②$^+$	0	575400	39700$^\Delta$
③$^+$	± 1750	544700	±39300
④	- 20540	291300	-40630
⑤	- 18050	672300	-64400

$^* P_{EQ} = P_{AX} + \frac{2M}{R}$ where $R = 29$ IN.

∴ THE CRITICAL LOADING CONDITION FOR SHELL BUCKLING IS CONDITION 5.

$P_{EQ} = 64,400$ lbs (ULT)

$^+$ USES EMPTY WEIGHTS

Δ ASSUMPTION OF SIMULTANEOUS LOADING (SEE PG 2) WOULD ONLY PRODUCE A P_{EQ} OF 52800 lbs WHICH IS LESS THAN CONDITION 5.

SHELL BUCKLING - ALUMINUM

$R = 29$ in $\quad E = 10 \times 10^6$ psi

TRY $t = .070$ in

$R/t = 414$

ANALYSIS REF "SEIDE, MORGAN et al" PG 39
(TRW REPORT EM 10-2?)

$C = .22$

$P_{cr} = 2\pi C E t^2$

$= 2\pi(.22)(10\times10^6)(.070)^2$

$= 67700$ lbs.

(BUCKLING) $M.S. = \frac{67700}{64400} - 1 = \underline{.05}$

SEPARATION SYSTEM

MAXIMUM TENSION LOAD ALONG SEPARATION
JOINT $= W$ lb/in

$$W = \frac{P}{2\pi R} + \frac{M}{\pi R^2} \times (.80)^*$$

$= .0055 P + .000303 M \quad$ for $R = 29$ in.

* FACTOR TO ACCOUNT FOR MODULUS EFFECT IN Mc/I DISTRIBUTION

SEPARATION SYSTEM (CONT)

$W_H = W \tan 20°$ (ASSUMES $\mu = 0$ & 20° RAMP ANGLE)

CONDITIONS 1 & 2 COMBINED ARE CRITICAL
(SEE NOTE PG 2)

$W = 247$ lb/in
$W_H = 90$ lb/in. } (ULT)

REQUIRED BAND LOAD, P_B

$P_B = 2 W_H R$
$= 2(90)(29) = 5220$ lbs

THIS IS THE LIMIT BAND LOAD (PRELOAD LEVEL).

$P_{B\,(ULT)} = 1.5\, P_B = 7830$ lbs. (ULT)

∴ A HEAO TYPE (ALSO M35) SEPARATION NUT SHOULD BE USED. ALSO USE THE HEAO SEPARATION BAND.

PRELOAD ALLOWABLE = 12500 lbs } REF CH7634
ULTIMATE ALLOWABLE > 20000 lbs } NUT ASSY SPEC.

(PRELOAD) $M.S. = \frac{12500}{5220} - 1 = \underline{1.39}$

(TENSION) $M.S. = \frac{20000}{7830} - 1 = \underline{1.55}$

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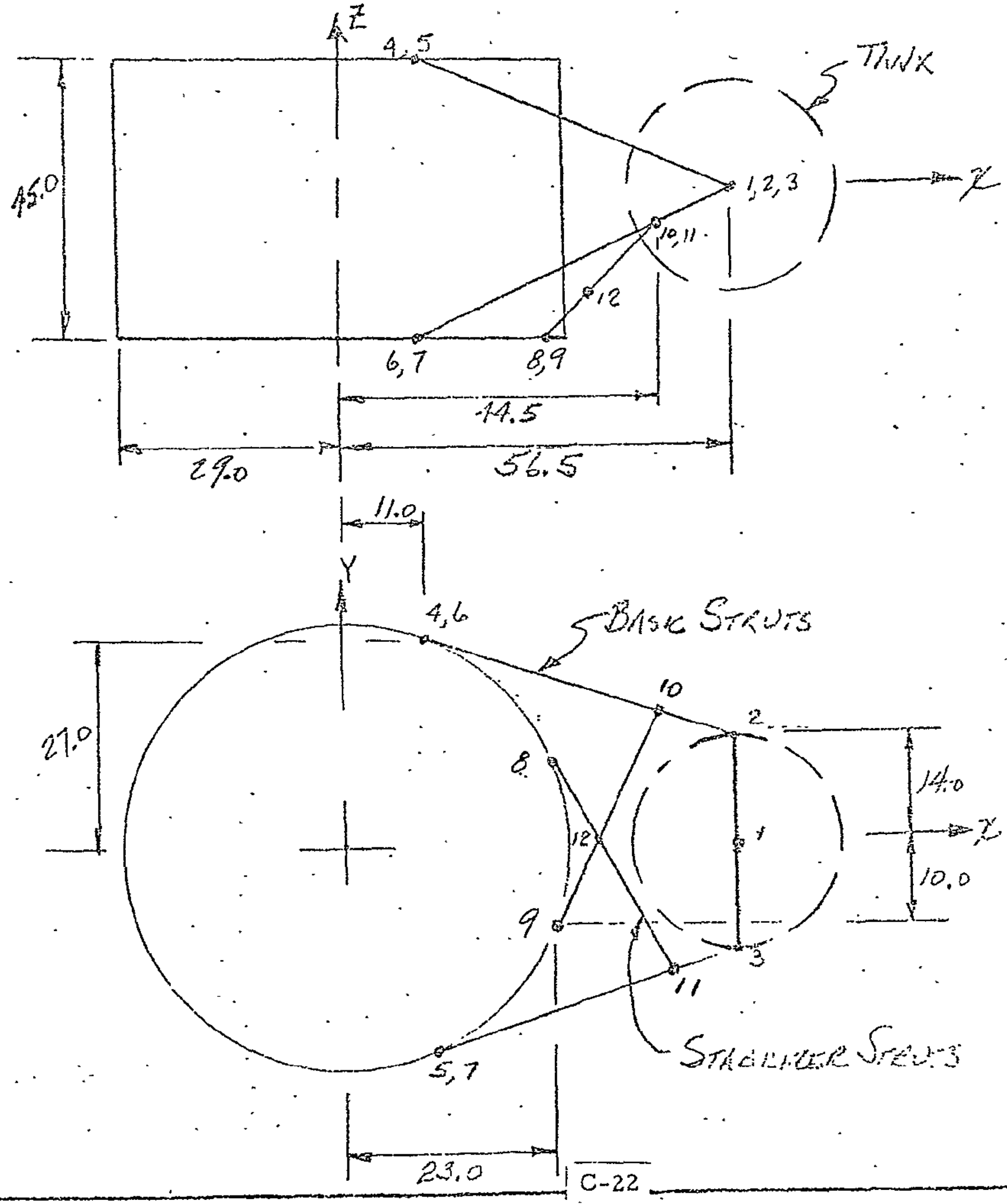
Tank Support Struts

Each Tank (Loaded) Weighs ≈ 400 lbs.



TANK STRUTS (CON'T)

BASIC STRUTS

3.0 IN. DIA × .025 IN. 6AL-4V TITANIUM

$A = .235 \text{ in}^2$

$I = .265 \text{ in}^4$

CRITICAL CONDITION : No. 5 REF PG 2

$P_{AX} = 1746$ lbs.

$M_{MAX} = 6817$ in-lbs (ULT)

$$f_b = \frac{P}{A} + \frac{Mc}{I} = \frac{1746}{.235} + \frac{6817(1.50)}{.265}$$

$= 46020$ psi (ULT)

$F_{cy} = 120000$ psi

$F_{ccr} = P_{cr}/A$

$R/t = 1.5/.025 = 60 \quad C = .38$ (REF TRW REPORT EM 10-26)

$P_{cr} = 2\pi C E t^2 = 25370$ lbs

$F_{ccr} = 25370/.235 = 107{,}950$ psi

$$M.S. = \frac{107950}{46020} - 1 = \underline{1.34}$$

Truss Struts (Cont)

Stabilizer Struts

2.0 in Dia × .025 in 6Al-4V Titanium

$A = .157\ in^2$

$I = .078\ in^4$

Critical Condition: No. 5 Ref Pg 2

$P_{AX} = 854$ lbs.
$M = 279$ in-lbs
(ULT)

$$f_b = \frac{P}{A} + \frac{Mc}{I} = \frac{854}{.157} + \frac{279(1.0)}{.078}$$

$= 9020$ psi (ULT)

$F_{cy} = 120000$ psi

$$M.S. = \frac{120000}{9020} - 1 = 11.\text{[illegible]}$$

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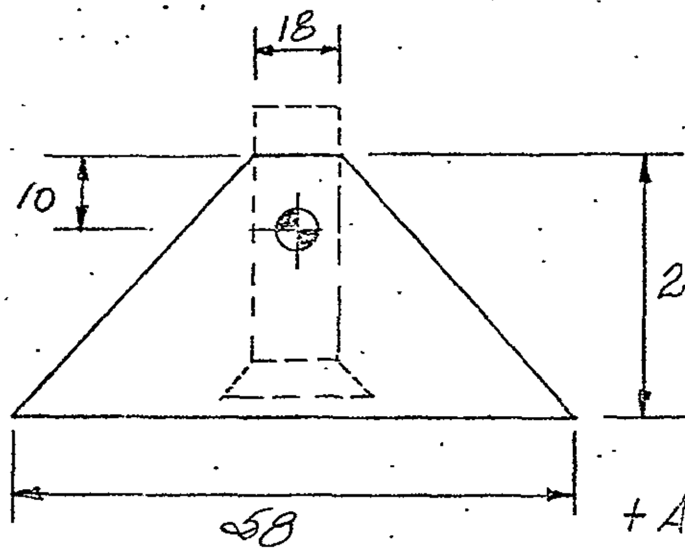
ENGINE SUPPORT CONE

ENGINE WEIGHT = 60 lbs.

MAXIMUM THRUST = 800 lbs. (LIMIT LOAD)



+AXIAL LOAD IS TENSION

LOADS @ TOP OF CONE - (ULTIMATE)

CONDITION		PAX	M	PLAT
1	CRASH	540	0	0
2	CRASH	0	2700	270
3	LANDING	± 72	2556	256
4	BOOST	-297	702	70
5	LIFT-OFF	-261	1620	162
6	ENGINE THRUST	1200	0	0

ENGINE SUPPORT CONE (CONT)

USING AN ALUMINUM CONE (6061-T6)

$E = 10 \times 10^6$ psi

$t = .025$ in.

ALLOWABLE BENDING MOMENT = 32000 IN-LBS
ALLOWABLE AXIAL COMPRESSION = 3500 lbs.*
ALLOWABLE SHEAR LOAD = 2800 lbs
} REF. TRW REPORT EM10-26

USING LOAD RATIOS:

R = APPLIED LOAD / ALLOWABLE LOAD

COND	R_{AX}	R_M	R_{LAT}	M.S. $\left(\frac{1}{\Sigma R} - 1\right)$
1	.154	0	0	HIGH
2	0	.084	.096	HIGH
3	.021	.080	.091	HIGH
4	.085	.022	.025	HIGH
5	.075	.051	.058	HIGH
6	.343	0	0	1.91

* RIVET BEARING, CRITICAL

WEIGHTS (SPACE STORABLE) - STRUCTURE

	WTS.
① CYLINDER - 1	
ALUMINUM t = .070 IN. L = 45 IN. D = 58 IN.	57.4
② SEPARATION SYSTEM - 1	
BANDS, SHOES, RETAINER, ORDNANCE (SAME AS HEAO)	35.0
③ SEPARATION I/F RINGS - 2	
ALUMINUM X-SECT AREA = .70 IN²	25.5
④ BOTTLE SUPPORT TRUSS - 4	
A) 4 STRUTS 3.0" x .025 (L = 61.8 IN) TITANIUM	39.5
B) 2 STRUTS 2" x .025 (L = 45.4 IN)	9.7
⑤ ENGINE SUPPORT CONE - 1	
ALUMINUM t = .025 IN.	8.1
Σ =	175.2
+20% (UNCERTAINTY PLUS FITTINGS) =	35.0
TOTAL =	210.2 lbs

TANK WEIGHTS

USING 300 psi TANKS

$R = 14$ IN

$f = \frac{PR}{2t}$

FOR $F_{TU} = 160000$ psi (6AL-4V TITANIUM)

$t = \frac{PR}{2F_{TU}}$ $P = 300 \times 2.22 = 666$ psi (BURST)

$= \frac{666(14)}{2(160000)}$

$= .029$ IN. (MIN) (USE $t = .033 \pm .004$)

SURFACE AREA = $4\pi R^2$

Weight = SURFACE $\times t \times w$

$= 4\pi(14)^2(.033)(.17)$

$= 13.8$ lbs

USE 18 lbs TO INCLUDE TEAR DROP, WELD ZONE & PORTS.

GROSS WEIGHTS

TOTAL WEIGHTS - (SPACE STORABLE)

① STRUCTURE	210.2 lbs
② PROPELLANT	1346 lbs.
③ TANKAGE (4 x 18 lbs)	72 lbs
④ PLUMBING	*
⑤ THERMAL INSULATION	*
⑥ HELIUM TANK & HELIUM 2% OF PROPELLANT (PER WHIT.)	27 lbs.
⑦ ENGINE	60 lbs
	1715 lbs
	+ CONTINGENCY

* NOT INCLUDED IN THIS PRELIMINARY ANALYSIS

$$\text{ALLOCATION} = \frac{\text{S/C GROSS} - \text{PAYLOAD}}{2}$$

$$= \frac{4145' - 750}{2} = \underline{1698 \text{ lbs}}$$

SAVINGS USING DIFFERENT STAGES

(ASSUMES UPPER STAGE IN STACK CAN USE LIGHTER STRUCTURE)

$P_{AX} = 4.35(1700 + 750) = 10660$ lbs (ULT)

$M = 83(2.70)(750) + 22.5(2.70)(1700)$

$= 271350$ in-lbs (ULT) @ BASE OF Upper Stg

Use $t = .050$ in

$P_{EQ} = P_{AX} + \frac{2M}{R} = 29400$ lbs.

$R = 29$ in.

$P_{cr} = 2\pi C E t^2$

$= 34560$ lbs.

M.S. = .17

Try $t = .045$

$P_{cr} = 28000$ lbs too low

Try $t = .048$

$P_{cr} = 31850$ lbs M.S. = .08

ΔWEIGHT

$$W_0 = \pi D t L w$$
$$= \pi(58)(.048)(45)(.10)$$
$$= 39.4 \text{ lbs}$$

$$\Delta W_1 = 57.4 - 39.4 = 18.0 \text{ lbs} \quad (\text{CYL})$$

$$\Delta W_2 = -\frac{39.4}{57.4}(25.5) + 25.5 = 8.0 \text{ lbs} \quad (\text{RING})$$

% SAVINGS FOR DIFFERENT
UPPER STAGE = 26.0 lbs

$$\% \text{ STRUCT SAVINGS} = \frac{26.0}{210.7} \times 100 = 12\%$$

$$\% \text{ MODULE SAVINGS} = \frac{26}{1700} \times 100 = 2\%$$

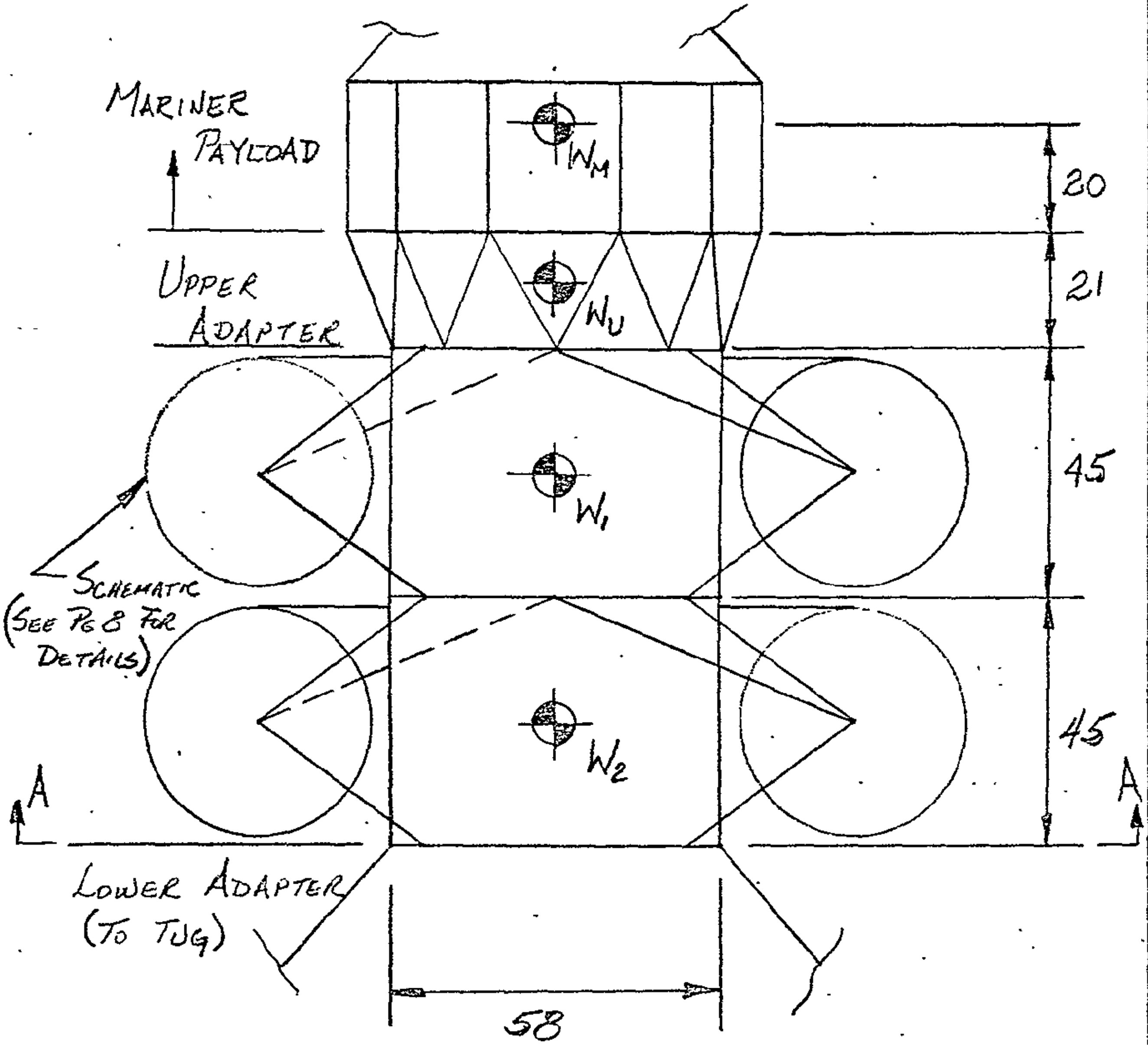
$$\% \text{ S/C GROSS SAVINGS} = \frac{26}{4150} \times 100 = 0.6\%$$

TANDEM MARINER ①

- 1210-lb_m MARINER S/C
- EARTH STORABLE PROPELLANT
- 800-Lb_f ENGINE

Tandem Module

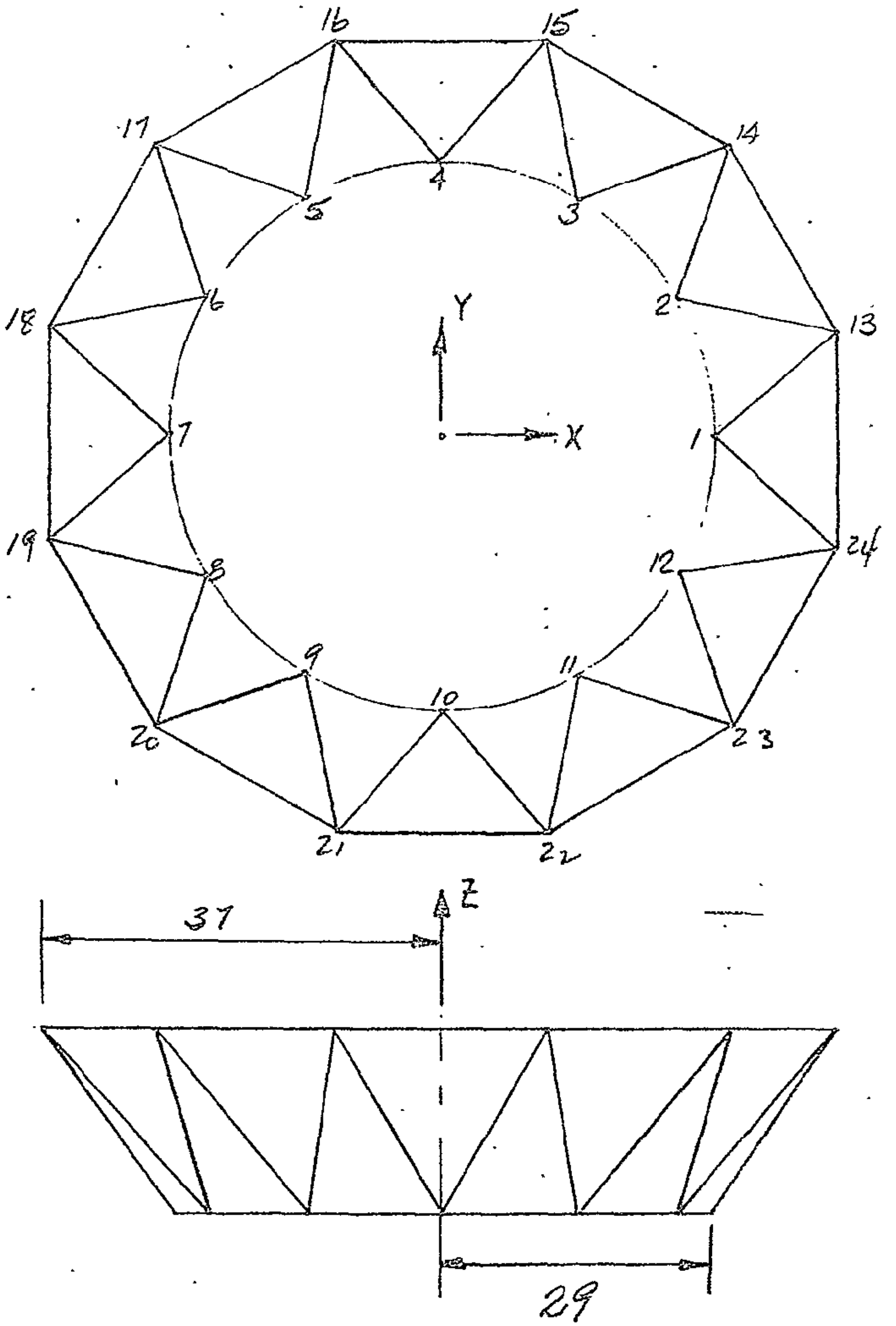
Earth Storable Propellant



C.G. Of Upper Adapter & 2 Modules Is At Their Geometric Centers

2

UPPER ADAPTER



PREPARED JAGLIKSMAN 3/5/75 REPORT NO.

CHECKED

MODEL MHC PMPO

MARINER PAYLOAD (1)

UPPER ADAPTER (CONT)

STRUTS : 2" × .058 6061-T6 ALUM TUBES

$A = .354 \text{ in}^2$

$I = .143 \text{ in}^4$

MAXIMUM LOAD = ±1802 lbs (ULT) COND 2 (Pg 4)

STRUT LENGTH = 24 in.

$$P_{cr} = \frac{\pi^2 EI}{L^2} = \frac{\pi^2 \times (10\times10^6)(.143)}{(24)^2} = 24500 \text{ lbs}$$

$$F_{cr} = P_{cr}/A = 69200 \text{ psi}$$

$$F_{cy} = 35000 \text{ psi}$$

$$f_c = P/A = 1802/.354 = 5100 \text{ psi (ULT)}$$

(COMPRESSION) $M.S. = \frac{35000}{5000} - 1 = \text{HIGH}$

WEIGHT OF UPPER ADAPTER

24 STRUTS AREA = .354 in L = 24 in $W_S = 20.4$

LOWER RING AREA = .70 in² $W_{LR} = 12.7$

UPPER RING* AREA = .20 in² $W_{UR} = 4.6$

TOTAL WEIGHT = 37.7 lbs

* ADDED TO EXISTING MARINER STRUCTURE

TANDEM MODULE - (EARTH STORABLE)

ASSUMING THAT THE MODULES ARE IDENTICAL, W1 = W2 & THE STRUCTURE IS THE SAME, THEN ONLY THE LOWER MODULE NEEDS ANALYSIS.

W_M = PAYLOAD WEIGHT = 1210 lbs.

W_U = UPPER ADAPTER WT = 38 lbs.

$W_1 = W_2$ = MODULE WEIGHT* = 5219 lbs.

*EACH MODULE CONTAINS 4314 lbs of PROPELLANT $\therefore W_1^* = W_2^*$ = 905 lbs EMPTY

CRITICAL LOADING CONDITIONS - (ULTIMATE)

CONFIGURATION	CONDITION	g_{AXIAL}	$g_{LATERAL}$
EMPTY	① CRASH Δ	9.00	0
EMPTY	② CRASH Δ	0	4.50
EMPTY	③ LANDING	±1.20	4.26
FULL	④ BOOST	-4.95	1.17
FULL	⑤ LIFT-OFF	-4.35	2.70

Δ NOTE: CONDITIONS ① & ② ARE NOW (4/16/75) ASSUMED TO ACT SIMULTANEOUSLY.

LOADS @ SECTION A-A - (ULTIMATE)

$$P_{AX} = g_{AX} \times \sum W_{TS}.$$

$$M = 131 \times g_L \times W_M + 100.5 \times g_L \times W_J + 67.5 \times g_L \times W_1 + 22.5 \times g_L \times W_2$$

CONDITION	P_{AX}	M	$P_{EQ.}^{*}$
(1)†	27520	0	27520 Δ
(2)†	0	1097000	75660 Δ
(3)†	± 3670	1038500	± 75290
(4)	- 51850	739500	-108850
(5)	- 50840	1706500	-168500

* $P_{EQ.} = P_{AX} + \dfrac{2M}{R}$ where $R = 29$ IN.

† USES EMPTY WEIGHTS

∴ THE CRITICAL LOADING CONDITION FOR SHELL BUCKLING IS CONDITION 5.

$P_{EQ.} = 168500$ lbs.

Δ ASSUMPTION OF SIMULTANEOUS LOADING (REF. PG 2) WOULD ONLY PRODUCE A $P_{EQ.}$ OF 103180 lbs WHICH IS LESS CRITICAL THAN CONDITION 5.

SHELL BUCKLING - ALUMINUM

$R = 29$ IN. $E = 10 \times 10^6$ psi

TRY $t = .110$ IN.

$R/t = 263$

ANALYSIS REF "SEIDE, MORGAN et al." PG 39
(TRW REPORT EM10-26)

$C = .26$

$$P_{cr} = 2\pi C E t^2$$
$$= 2\pi (.26)(10 \times 10^6)(.110)^2$$
$$= 197600 \text{ lbs.}$$

(BUCKLING) $M.S. = \frac{197600}{168500} - 1 = \underline{.17}$

SEPARATION SYSTEM

MAXIMUM TENSION LOAD ALONG SEPARATION

JOINT = W lb/IN

$$W = \frac{P}{2\pi R} + \frac{M}{\pi R^2} \times (.80)^*$$

$$= .0055P + .000303\,M \quad \text{for } R = 29\text{ IN.}$$

* FACTOR TO ACCOUNT FOR MODULUS EFFECT IN Mc/I DISTRIBUTION.

SEPARATION SYSTEM (CONT)

$W_H = W \tan 20°$ (ASSUMES $\mu = 0$ & 20° RAMP ANGLE)

CONDITIONS 1 & 2 COMBINED ARE CRITICAL

$$\left.\begin{aligned} W &= 484 \text{ lb/in} \\ W_H &= 176 \text{ lb/in} \end{aligned}\right\} \text{(ULT)}$$

REQUIRED BAND LOAD, P_B

$$P_B = 2 W_H R$$
$$= 2(176)(29) = 10200 \text{ lbs}$$

THIS IS THE LIMIT BAND LOAD (PRELOAD LEVEL)

$$P_{B\,(ULT)} = 1.5 P_B = 15300 \text{ lbs (ULT)}$$

∴ A HEAO TYPE (ALSO M35) SEPARATION NUT SHOULD BE USED. ALSO USE THE HEAO SEPARATION RINGS & SEPARATION BAND.

PRELOAD ALLOWABLE = 12500 lbs. } REF. C117634
ULTIMATE ALLOWABLE = 20000 lbs } NUT ASSY. SPEC.

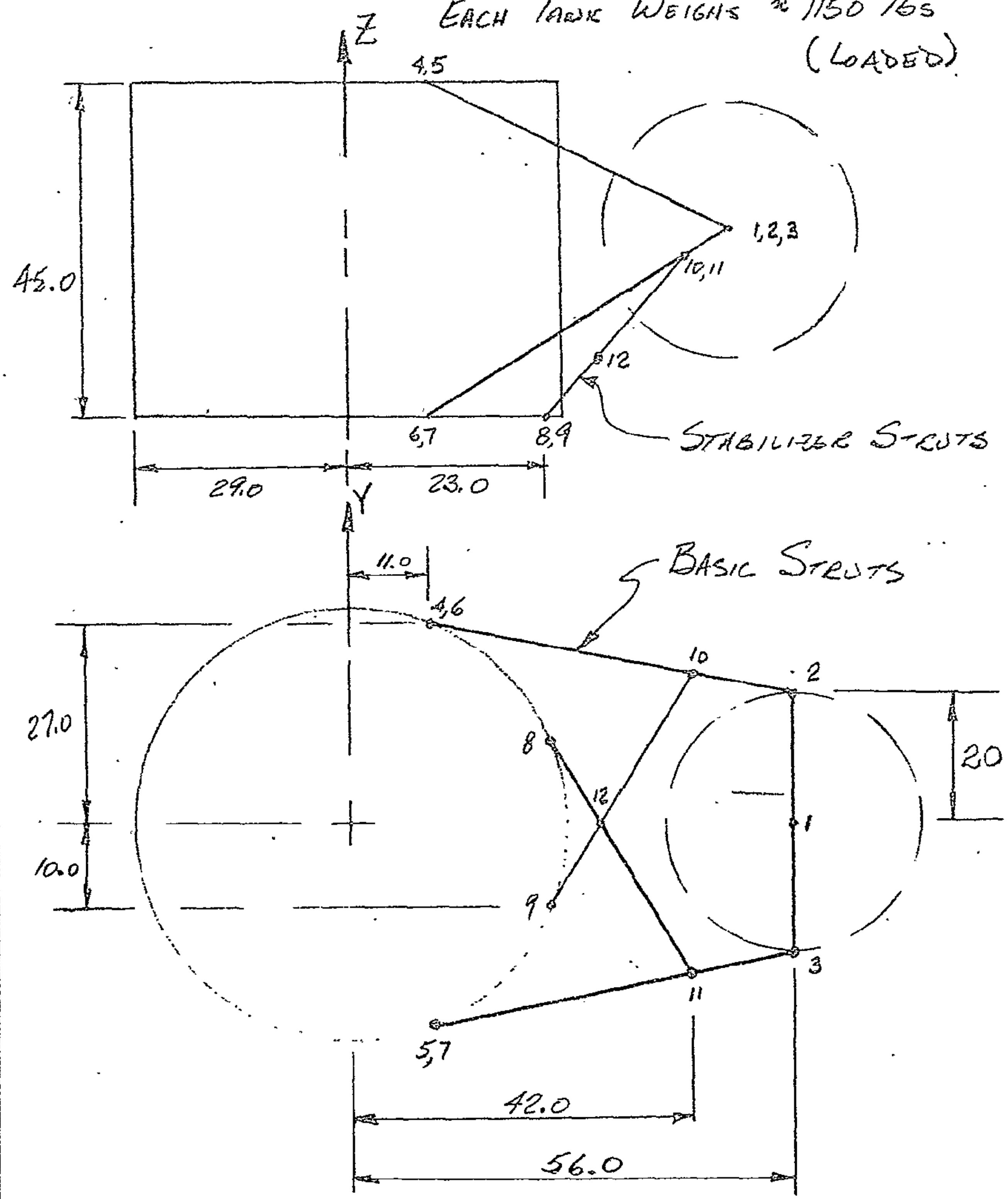
$$\text{(PRELOAD) M.S.} = \frac{12500}{10200} - 1 = \underline{\underline{.22}}$$

$$\text{(TENSION) M.S.} = \frac{20000}{15300} - 1 = \underline{\underline{.30}}$$

MARINER PAYLOAD ①

TANK SUPPORT STRUTS

EACH TANK WEIGHS ≈ 1150 lbs (LOADED)



SYSTEMS 1441 REV. 8-67

TANK SUPPORT STRUTS

THE BASIC STRUTS ARE 3.0 INCH DIAMETER, .060 INCH WALL 6AL-4V TITANIUM.

$A = .554 \text{ in}^2$

$I = .599 \text{ in}^4$

CRITICAL CONDITION : NO. 5 REF PG 4

$$\left.\begin{aligned} P_{AX} &= 4071 \text{ lbs} \\ M_{MAX} &= 24603 \text{ in-lbs} \end{aligned}\right\} \text{(ULT)}$$

MEMBER 11-7

$$f_b = \frac{P}{A} + \frac{Mc}{I} = \frac{4071}{.554} + \frac{24603\,(1.5)}{.599}$$

$= 66300$ psi

$F_{cy} = 120000$ psi

$$M.S. = \frac{120000}{66300} - 1 = \underline{.80}$$

SYSTEMS 1441 REV. 8-67

ENGINE SUPPORT CONE

THIS CONE SUPPORTS A SIMILAR ENGINE TO THE PIONEER PAYLOAD ENGINE.

WEIGHT = 60 lbs.

THRUST = 800 lbs (LIMIT LOAD)

∴ THE SAME CONE CAN BE USED

MATL: ALUMINUM (6061-T6)

E : 10×10^6 psi

t : .025

(MIN.) M.S. = 1.91

(11)

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MODEL MMCPHPO

MARINER PAYLOAD (1)

WEIGHTS - (EARTH STORABLE) - STRUCTURE

	WTS
(1) CYLINDER - 1	
ALUMINUM $t = .110$ IN. $L = 45$ IN $D = 58$ IN	90.2
(2) SEPARATION SYSTEM - 1	
° BANDS, SHOES, RETAINER, ORDNANCE (SAME AS HEAD)	35.0
(3) SEPARATION I/F RINGS - 2	
ALUMINUM X-SECT AREA = 1.00 IN^2	36.4
(4) BOTTLE SUPPORT TRUSS - 1	
A) 4 STRUTS 2.5" × .030 $L = 50.8$ (TITANIUM)	76.5
B) 2 STRUTS 2.0" × .025 $L = 41.2$ (TITANIUM)	13.8
(5) ENGINE SUPPORT CONE - 1	
ALUMINUM $t = .025$ IN.	8.1
Σ =	260.0
+20% (UNCERTAINTY PLUS FITTINGS) =	52.0
TOTAL =	312.0 lbs

SYSTEMS 1441 REV. 8-67

TANK WEIGHTS

USING 300 psi TANKS

$R = 20$ IN.

$$f = \frac{\rho R}{2t}$$

FOR $F_{TU} = 160,000$ psi (6Al-4V TITANIUM)

$$t = \frac{\rho R}{2F_{TU}} \qquad \rho = 300 \times 2.22 = 666 \text{ psi (BURST)}$$

$$= \frac{666(20)}{2(160,000)}$$

$= .042$ IN. (MIN) (USE $t = .046 \pm .004$)

SURFACE AREA $= 4\pi R^2$

WEIGHT = SURFACE $\times t \times \omega$

$= 4\pi(20)^2 \times .046 \times .17$

$= 39.3$ lbs

USE 48 lbs TO INCLUDE TEAR DROP, WELD ZONE & PORTS

TOTAL WEIGHTS - (EARTH STORABLE)

① STRUCTURE	312	lbs
② PROPELLANT	4314	lbs
③ TANKAGE (4 × 48 lbs)	192	lbs.
④ PLUMBING	*	lbs
⑤ THERMAL INSULATION	*	lbs.
⑥ HELIUM TANK & HELIUM (2% OF PROP)	86.3	lbs
⑦ ENGINE	60	lbs
	4964	lbs.

* NOT INCLUDED IN THIS PRELIMINARY ANALYSIS

$$\text{ALLOCATION} = \frac{\text{S/C GROSS} - \text{PAYLOAD} - \text{ADAPTER (UPPER)} + \text{CONTINGENCY}}{2}$$

$$= \frac{11686 - 1210 - 38}{2}$$

$$= \underline{\underline{5219 \text{ lbs}}}$$

TANDEM MARINER ②

- 1210-lbm MARINER S/C
- SPACE STORABLE PROPELLANT
- 800-lbf ENGINE

(1)

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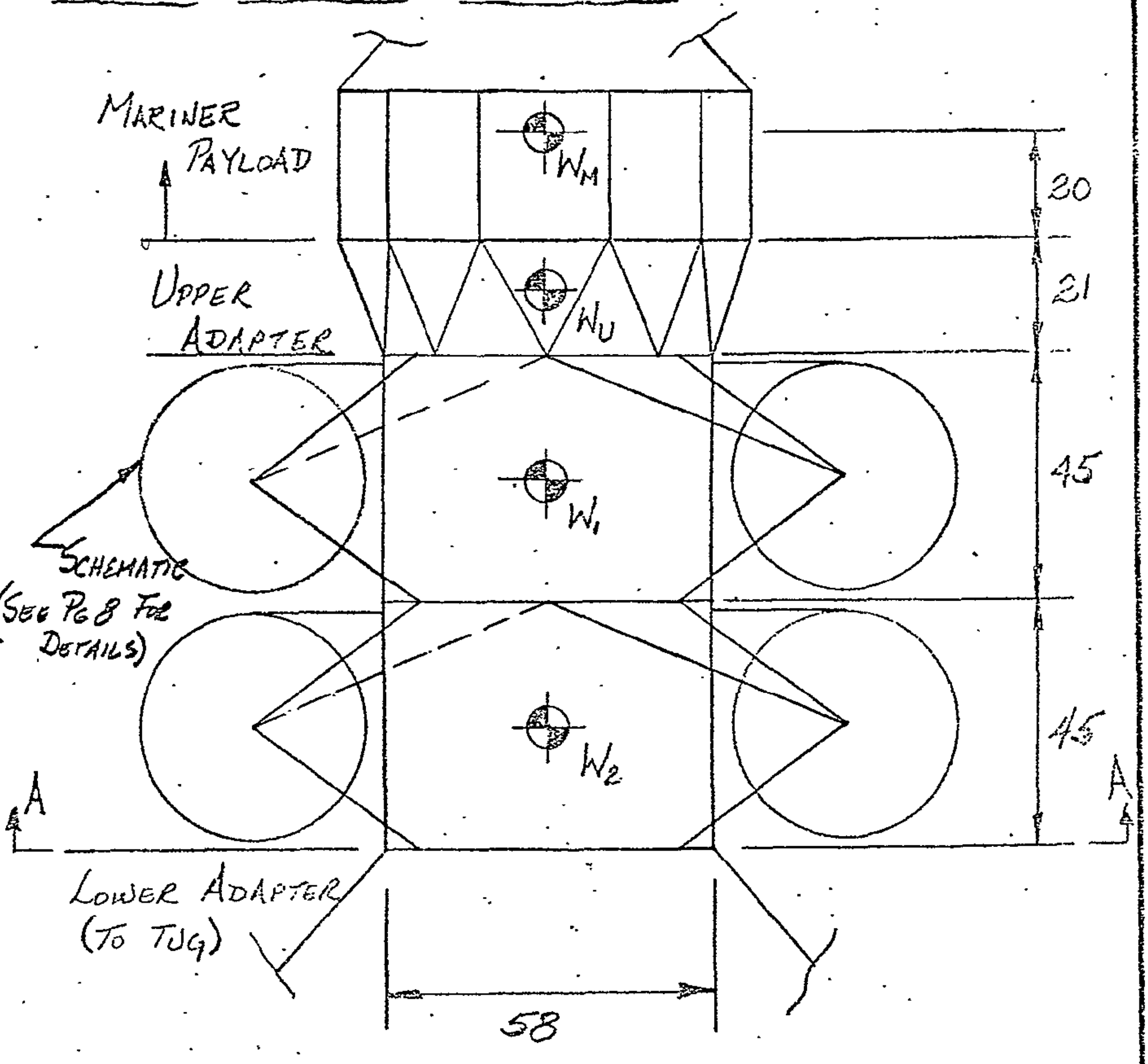
MODEL MMC PMPO

MARINER PAYLOAD (2)

TANDEM MODULE

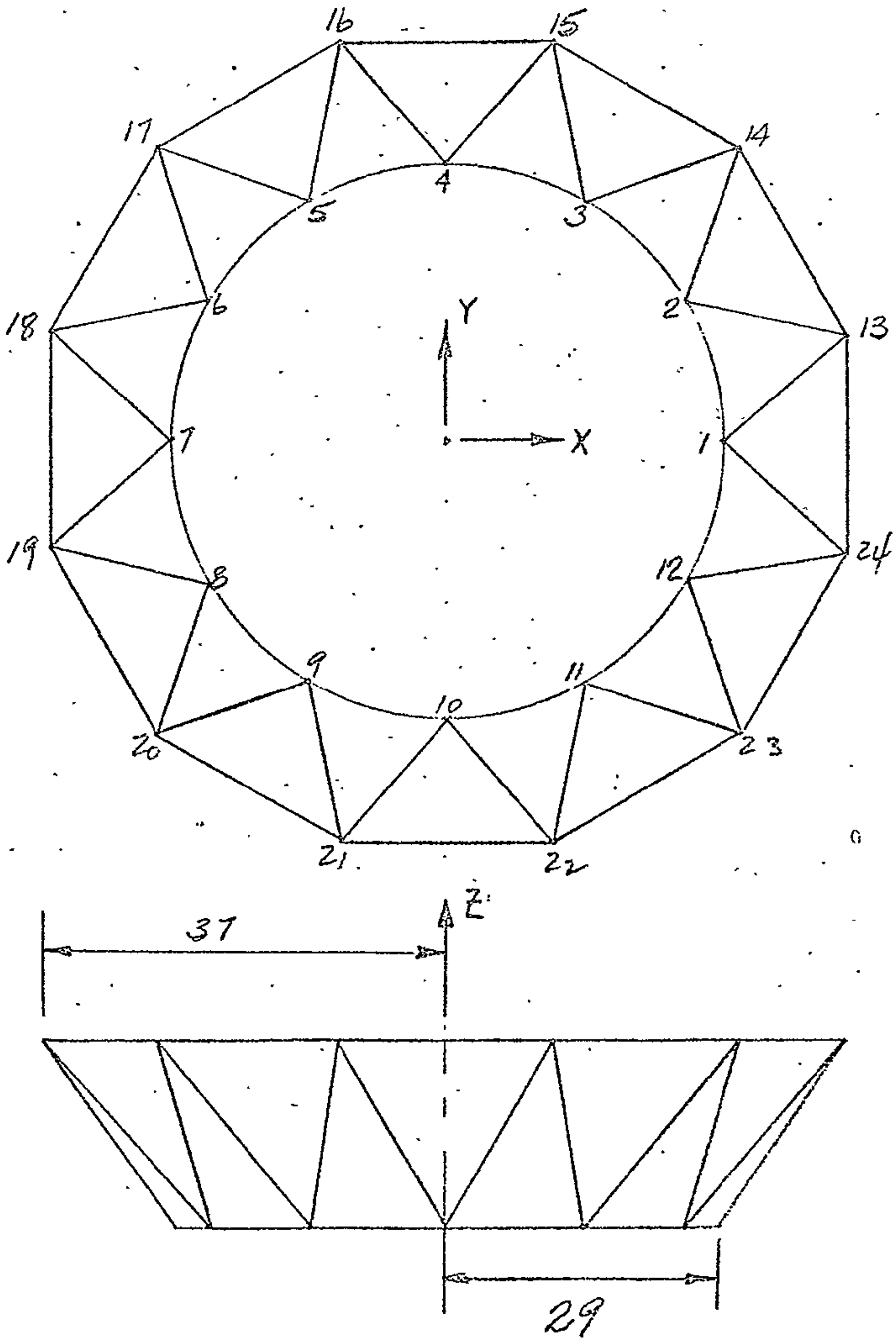
SPACE STORABLE PROPELLANT



C.G. OF UPPER ADAPTER & 2 MODULES IS AT THEIR GEOMETRIC CENTERS

SYSTEMS 1441 REV. 8-67

UPPER ADAPTER



UPPER ADAPTER (CONT)

STRUTS : 2" × .058 6061-T6 ALUM TUBES
$A = .354 \text{ in}^2$
$I = .143 \text{ in}^4$

MAXIMUM LOAD = ±1802 lbs (ULT) COND 2 (Pg 4)
STRUT LENGTH = 24 in.

$$P_{cr} = \frac{\pi^2 E I}{L^2} = \frac{\pi^2 \times (10 \times 10^6)(.143)}{(24)^2} = 24500 \text{ lbs}$$

$$F_{cr} = P_{cr}/A = 69200 \text{ psi}$$

$$F_{cy} = 35000 \text{ psi}$$

$$f_c = P/A = 1802/.354 = 5100 \text{ psi (ULT)}$$

(COMPRESSION) $M.S. = \frac{35000}{5000} - 1 =$ HIGH

WEIGHT OF UPPER ADAPTER

24 STRUTS	AREA = .354 in L = 24 in	W_S = 20.4
LOWER RING	AREA = .70 in²	W_{LR} = 12.7
UPPER RING*	AREA = .20 in²	W_{UR} = 4.6

TOTAL WEIGHT = 37.7 lbs

* ADDED TO EXISTING MARINER STRUCTURE

MARINER PAYLOAD ②

TANDEM MODULE - (SPACE STORABLE)

ASSUMING THAT THE MODULES ARE IDENTICAL, W1 = W2 & THE STRUCTURE IS THE SAME, THEN ONLY THE LOWER MODULE NEEDS ANALYSIS.

W_M = PAYLOAD WEIGHT = 1210 lbs.

W_U = UPPER ADAPTER WT = 38 lbs.

$W_1 = W_2$ = MODULE WEIGHT* = 2676 lbs.

*EACH MODULE CONTAINS 2180 lbs. of PROPELLANT ∴ $W_1^* = W_2^*$ = 496 lbs EMPTY

CRITICAL LOADING CONDITIONS - (ULTIMATE)

CONFIGURATION	CONDITION	g_{AXIAL}	$g_{LATERAL}$
EMPTY	① CRASH Δ	9.00	0
EMPTY	② CRASH Δ	0	4.50
EMPTY	③ LANDING	±1.20	4.26
FULL	④ BOOST	-4.95	1.17
FULL	⑤ LIFT-OFF	-4.35	2.70

Δ NOTE: CONDITIONS ① & ② ARE NOW (4/16/75) ASSUMED TO ACT SIMULTANEOUSLY.

MARINER PAYLOAD ②

LOADS @ SECTION A-A' - (ULTIMATE)

$$P_{AX} = g_{AX} \times \Sigma \text{ WTS.}$$

$$M = 131 \times g_L \times W_M + 100.5 \times g_L \times W_J + 67.5 \times g_L \times W_1 + 22.5 \times g_L \times W_2$$

CONDITION	P_{AX}	M	$P_{EQ.}$*
①†	20160	0	20160 Δ
②†	0	931400	64200 Δ
③†	± 2690	881700	±63500
④	- 32670	471700	-65200
⑤	- 28710	1088600	-103800

$$^*P_{EQ.} = P_{AX} + \frac{2M}{R} \quad \text{Where } R = 29 \text{ IN.}$$

† USES EMPTY WEIGHTS

∴ THE CRITICAL LOADING CONDITION FOR SHELL BUCKLING IS CONDITION 5.

$$P_{EQ.} = 103800 \text{ lbs.}$$

Δ ASSUMPTION OF SIMULTANEOUS LOADING (REF. PG 2) WOULD ONLY PRODUCE A $P_{EQ.}$ OF 84360 lbs WHICH IS LESS CRITICAL THAN CONDITION 5.

SHELL BUCKLING - ALUMINUM

$R = 29$ IN. $E = 10 \times 10^6$ psi

TRY $t = .090$ IN.

$R/t = 322$

ANALYSIS REF "SEIDE, MORGAN et al" PG 39
(TRW REPORT EM 10-26)

$C = .24$

$P_{cr} = 2\pi C E t^2$

$= 2\pi(.24)(10\times10^6)(.090)^2$

$= 122000$ lbs.

(BUCKLING) $M.S. = \frac{122000}{103800} - 1 = .17$

SEPARATION SYSTEM

MAXIMUM TENSION LOAD ALONG SEPARATION

JOINT = w lb/in

$$w = \frac{P}{2\pi R} + \frac{M}{\pi R^2} \times (.80)^*$$

$= .0055P + .000303M$ for $R = 29$ in.

* FACTOR TO ACCOUNT FOR MEDDLES EFFECT IN Mc/I DISTRIBUTION.

SYSTEMS 1441 REV. 8-67

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SEPARATION SYSTEM (CONT)

$W_H = W \tan 20°$ (ASSUMES $\mu = 0$ & 20° RAMP ANGLE)

CONDITIONS 1 & 2 COMBINED ARE CRITICAL

$W = 393$ lb/in
$W_H = 143$ lb/in } (ULT)

REQUIRED BAND LOAD, P_B

$P_B = 2 W_H R$
$= 2(143)(29) = 8300$ lbs

THIS IS THE LIMIT BAND LOAD (PRELOAD LEVEL)

$P_{B\,(ULT)} = 1.5 P_B = 12450$ lbs (ULT)

∴ A HEAO TYPE (ALSO M35) SEPARATION NUT SHOULD BE USED. ALSO USE THE HEAO SEPARATION RINGS & SEPARATION BAND.

PRELOAD ALLOWABLE = 12500 lbs } REF. C117634
ULTIMATE ALLOWABLE = 20000 lbs } NUT ASSY. SPEC.

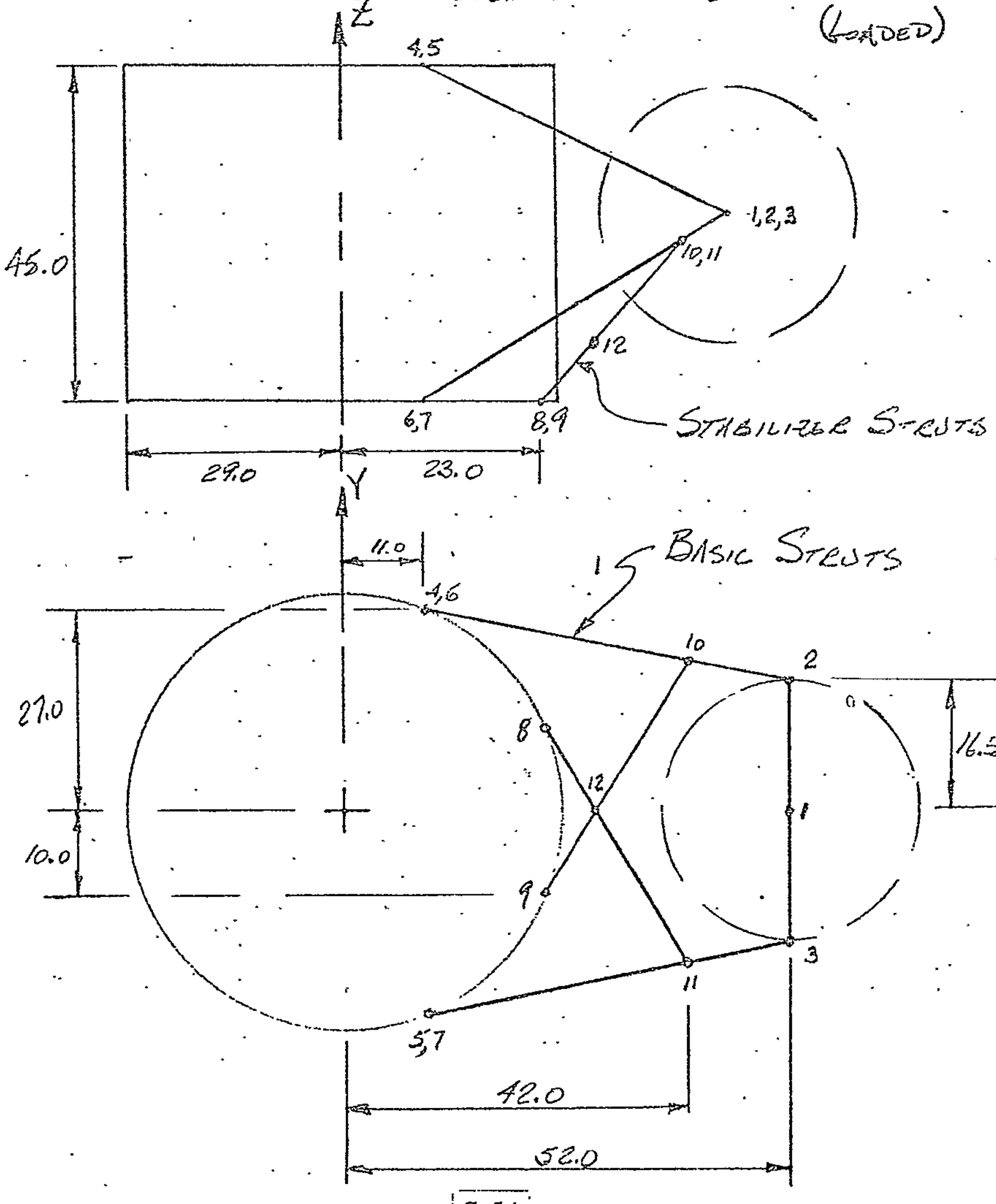
(PRELOAD) $M.S. = \frac{12500}{8300} - 1 = \underline{.50}$

(TENSION) $M.S. = \frac{20000}{12450} - 1 = \underline{.60}$

MARINER PAYLOAD ②

TANK SUPPORT STRUTS

EACH TANK WEIGHS ≈ 600 lbs (LOADED)



MARINER PAYLOAD ②

TANK SUPPORT STRUTS

THE BASIC STRUTS ARE 3.0 INCH DIAMETER, .025 INCH WALL 6AL-4V TITANIUM.

$A = .235 \ in^2$

$I = .265 \ in^4$

CRITICAL CONDITION: NO. 5 REF PG 4

$$\left.\begin{array}{l} P_{AX} = 653 \text{ lbs} \\ M_{MAX} = 8753 \text{ in-lbs} \end{array}\right\} (\text{ULT}) \quad \text{MEMBER } 11\text{-}7$$

$$f_b = \frac{P}{A} + \frac{Mc}{I} = \frac{653}{.235} + \frac{8753\,(1.50)}{.265}$$

$= 52300$ psi

$F_{cy} = 120000$ psi

$F_{ccr} = P_{cr}/A$

$R/t = 1.5/.025 = 60 \qquad C = .38$ (REF TRW REPORT EM10-26)

$P_{cr} = 2\pi C E t^2 = 25370$ lbs.

$F_{ccr} = 25370/.235 = 107,950$ psi

$$M.S. = \frac{107950}{52300} - 1 = \underline{1.06}$$

MARINER PAYLOAD ②

ENGINE SUPPORT CONE

THIS CONE SUPPORTS A SIMILAR ENGINE TO THE PIONEER PAYLOAD ENGINE.

WEIGHT = 60 lbs.

THRUST = 800 lbs (LIMIT LOAD)

∴ THE SAME CONE CAN BE USED

MATL: ALUMINUM (6061-T6)

E : 10×10^6 psi

t : .025

(MIN.) M.S. = 1.91

(11)

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MODEL MMCPMPO

MARINER PAYLOAD ②

WEIGHTS - (SPACE STORABLE) - STRUCTURE

	WTS
① CYLINDER - 1	
ALUMINUM t=.090 IN. L=45 IN D=58 IN	73.8
② SEPARATION SYSTEM - 1	
BANDS, SHOES, RETAINER, ORDNANCE (SAME AS HEAO)	35.0
③ SEPARATION I/F RINGS - 2	
ALUMINUM X-SECT AREA = .90 IN²	32.8
④ BOTTLE SUPPORT TRUSS - 4	
A) 4 STRUTS 2.5"×.030 L=48.0 (TITANIUM)	30.7
B) 2 STRUTS 2.0"×.025 L=39.4 (TITANIUM)	8.4
⑤ ENGINE SUPPORT CONE	
ALUMINUM t=.025 IN.	8.1
Σ =	188.8
+20% (UNCERTAINTY PLUS FITTINGS) =	37.8
TOTAL =	226.6 lb

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SYSTEMS 1441 REV. 8-67

TANK WEIGHTS

USING 300 psi TANKS

$R = 17$ IN.

$$f = \frac{pR}{2t}$$

FOR $F_{TU} = 160{,}000$ psi (6AL-4V TITANIUM)

$$t = \frac{pR}{2F_{TU}} \qquad p = 300 \times 2.22 = 666 \text{ psi (BURST)}$$

$$= \frac{666(17)}{2(160{,}000)}$$

$$= .036 \text{ IN. (MIN)} \quad (\text{USE } t = .040 \pm .004)$$

SURFACE AREA $= 4\pi R^2$

WEIGHT = SURFACE $\times t \times w$

$$= 4\pi(17)^2 \times .040 \times .17$$

$$= 24.7 \text{ lbs}$$

USE 30 lbs TO INCLUDE TEAR DROP, WELD ZONE & PORTS

GROSS WEIGHTS

TOTAL WEIGHTS - (SPACE STORABLE)

1 STRUCTURE	226.6	lbs
2 PROPELLANT	2180	lbs
3 TANKAGE (4 x 30 lbs)	120	lbs
4 PLUMBING	*	lbs
5 THERMAL INSULATION	*	lbs
6 HELIUM TANK & HELIUM (2% OF PROP)	43.6	lbs
7 ENGINE	60	lbs
	2630	lbs

* NOT INCLUDED IN THIS PRELIMINARY ANALYSIS

+ CONTINGENCY

$$\text{ALLOCATION} = \frac{\text{S/C GROSS} - \text{PAYLOAD} - \text{ADAPTER (UPPER)}}{2}$$

$$= \frac{6600 - 1210 - 38}{2}$$

$$= 2676 \text{ lbs}$$

APPENDIX D

PERFORMANCE DATA FOR VARIOUS SHUTTLE/UPPER STAGE COMBINATIONS

This appendix presents supporting data on the performance of various Space Shuttle/upper stage combinations that were designated by NASA for purposes of this study as launch vehicle candidates (see Section 7 of Volume II).

1. CANDIDATE LAUNCH VEHICLES

The launch vehicle combinations considered include the following twenty:

Shuttle Upper Stage	Kick Stage
Centaur D-1S (planetary)	
1. Centaur D-1S	Burner II (2300)
2. Centaur D-1S	TE 364-4 (2300), spin-stabilized
3. Centaur D-1S	APM-I
4. Centaur D-1S	APM-1, spin-stabilized
5. Centaur D-1S	PM (2300)
6. Centaur D-1S (plus spin table)	
7. Centaur D-1S	
Space Tug	
8. Space Tug	Burner II (2300)
9. Space Tug	TE 364-4 (2300), spin-stabilized
10. Space Tug	APM-I
11. Space Tug	APM-I, spin-stabilized
12. Space Tug	PM (2300)
13. Space Tug (plus spin table)	
14. Space Tug	
Transtage	
15. Dual short Transtage	Kick stage (4400)
16. Dual short transtage	Kick stage (4400), spin-stabilized

17. Dual short Transtage

18. Dual short Transtage (plus spin table)

In addition, the performance of two Titan III-class expendable launch vehicles was considered (for comparison only):

Titan IIIE/Centaur D-1T	Burner II (2300)
Titan IIIE/Centaur D-1T	TE 364-4 (2300), spin-stabilized.

Note that the designations used above are not firmly established. Numbers in parentheses following the designation of the kick stage indicate the propellant loading (in lb_m). The kick stage denoted as APM-I, currently in advanced design, was formerly designated as SPM (1800), where 1800 is the propellant mass plus motor case (in kg).

The term SPM (1800) was used consistently in the body of this report (Volume II). Performance data for the first 14 upper stage combinations listed above were generated by TRW. The data on the final 6 combinations were reproduced from external sources.

2. PERFORMANCE CHARACTERISTICS

A detailed and precise launch phase trajectory simulation was performed taking all velocity losses into account. The following 16 charts with 5 columns of entries defined as follows, give performance detail:

Column 1: Twice the total vehicle energy at kick stage burnout, C_3

Column 2: Net launch vehicle payload (injected mass)

Column 3: Vehicle mass at first burn ignition. This column will show where off-loading of the upper stage begins (if required)

Column 4: Total gravity loss of injection maneuver (both stages) is defined by

$$G_i = \Delta V_{RC_i} - \Delta V_{IMP_i}$$

where ΔV_{RC} = the ideal stage ΔV capability as computed from the rocket equation, and ΔV_{IMP} = the propulsive ΔV that must be added to the upper stage ignition speed in order to increase the total vehicle energy to the actual stage burnout energy.

Column 5: Total vehicle ΔV. This is the difference between the vehicle speed at kick stage burnout and at upper stage ignition.

Above the tabular data several additional lines of information are printed out. The first line gives the spin-table mass (zero for three-axis stabilized payloads). The second line identifies the upper stage and its mass and performance parameters (in the order slated):

1) Usable fuel mass (kg)

2) Burnout mass (kg)

3) Adapter mass (kg)

4) Nonimpulsive inert mass (kg)

5) Specific impulse (sec)

6) Thrust magnitude (lb)

7) Maximum allowable first ignition mass

The third line (if present) identifies the kick stage and its characteristics (items (1) through (6)).

Figures D-1 to D-3 present the launch vehicle performance characteristics in terms of net payload mass versus injection C_3 (columns 1 and 2 of the tabulated data).

The performance of single upper stages (Centaur class and Space Tug) are compared in Figures D-4 and D-5. These stages are considered for use in the Mercury orbiter mission only, where the required C_3 values are so low as to make the addition of a solid kick stage unnecessary.

Assumptions used in simulating the mass characteristics, specific impulse and thrust levels of the various vehicles are summarized as follows:

a) First burn ignition occurs in a circular earth orbit at 160 km altitude. The earth is assumed to be a sphere with a radius of 20,925,673 feet (6,378.222 km).

b) The thrust and specific impulse of both upper stage and kick stage are assumed to be specified constants as given in Table D-1.

SPINTABLE MASS	0.0						
SPACE-TUG(EXPEN)	22625.0	2642.0	104.5	109.7	456.5	15000.0	28377.0
APM-I	1710.0	144.0	9.0	0.0	297.0	15000.0	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
159.832	1000.000	28376.999	521.712	7381.025
146.501	1200.000	28376.999	500.278	6965.897
135.325	1400.000	28376.999	479.815	6613.490
125.706	1600.000	28376.999	460.216	6307.414
117.263	1800.000	28376.999	441.403	6036.970
109.737	2000.000	28376.999	423.316	5794.804
102.946	2200.000	28376.999	405.986	5575.656
96.757	2400.000	28376.999	389.134	5375.637
91.070	2600.000	28376.999	372.966	5191.791
85.808	2800.000	28376.999	357.373	5021.818
80.91[illegible]	3000.000	28376.999	342.328	4863.89[illegible]
76.329	3200.000	28376.999	327.808	4716.54[illegible]
72.025	3400.000	28376.999	313.794	4578.565
67.966	3600.000	28376.999	300.265	4448.950
64.125	3800.000	28376.999	287.264	4326.852
60.479	4000.000	28376.999	274.596	4211.554
57.010	4200.000	28376.999	262.425	4102.437
53.700	4400.000	28376.999	250.678	3998.96[illegible]
50.536	4600.000	28376.999	239.342	3900.669
47.506	4800.000	28376.999	228.403	3807.135
44.597	5000.000	28376.999	217.851	3717.997
41.802	5200.000	28376.999	207.674	3632.926
39.11[illegible]	5400.000	28376.999	197.862	3551.628
36.516	5600.000	28376.999	188.405	3473.836
34.012	5800.000	28376.999	179.292	3399.307
31.591	6000.000	28376.999	170.515	3327.819
29.25	6200.000	28376.999	162.[illegible]64	3259.173

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SPINTABLE MASS	[illegible]						
SPACE-TUG(EXPEN)	22625.0	2642.0	104.5	109.7	456.5	15000.0	28377.0
BII(2300)	1043.3	226.[illegible]	12.7	11.2	283.[illegible]	150[illegible]	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
156.156	800.000	27624.399	555.735	7275.103
143.915	1000.000	27824.399	542.390	6878.243
133.862	1200.000	28044.399	529.765	6546.371
125.337	1400.000	28264.399	517.767	6262.626
117.566	1600.000	28376.999	501.772	6004.49[illegible]
110.284	1800.000	28376.999	481.711	5779.510
103.700	2000.000	28376.999	462.385	5559.547
97.686	2200.000	28376.999	443.752	5364.739
92.144	2400.000	28376.999	425.778	5186.217
87.003	2600.000	28376.999	408.432	5020.775
82.205	2800.000	28376.999	391.686	4866.687
77.705	3000.000	28376.999	375.515	4722.571
73.467	3200.000	28376.999	359.896	4587.303
69.462	3400.000	28376.999	344.809	4459.943
65.663	3600.000	28376.999	330.234	4339.72[illegible]
62.051	3800.000	28376.999	316.154	4225.965
58.608	4000.000	28376.999	302.552	4118.110
55.317	4200.000	28376.999	289.412	4015.66[illegible]
52.167	4400.000	28376.999	276.72[illegible]	3915.18[illegible]
49.145	4600.000	28376.999	264.[illegible]63	3825.294
46.242	4800.000	28376.999	252.627	3736.65[illegible]
43.448	5000.000	28376.999	241.200	3651.962
40.755	5200.000	28376.999	230.171	3570.94[illegible]
38.157	5400.000	28376.999	219.528	3493.343
35.647	5600.000	28376.999	209.260	3418.935
33.219	5800.000	28376.999	199.358	3347.513
30.868	6000.000	28376.999	189.812	3278.896
28.590	6200.000	28376.999	180.612	3212.892

SPINTABLE MASS 0.0
SPACE-TUG(EXPEN) 22625.0 2642.0 164.5 1[illegible]9.7 456.5 15[illegible] [illegible] 28377.0
PM(23[illegible]) 1043.0 175.0 14.0 0.0 285.0 15000.0

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
159.774	800.000	27543.199	559.547	7390.625
146.837	1000.000	27763.199	545.991	6973.479
136.315	1200.000	27983.199	533.184	6627.622
127.451	1400.000	28203.199	521.024	6331.648
119.634	1600.000	28376.999	507.470	6070.428
112.147	1800.000	28376.999	487.200	5830.078
105.398	2000.000	28376.999	467.676	5612.907
99.247	2200.000	28376.999	448.855	5414.748
93.591	2400.000	28376.999	430.703	5232.518
88.351	2600.000	28376.999	413.186	5063.869
83.468	2800.000	28376.999	396.277	4906.973
78.894	3000.000	28376.999	379.949	4760.370
74.590	3200.000	28376.999	364.179	4622.878
70.526	3400.000	28376.999	348.947	4493.522
66.675	3600.000	28376.999	334.232	4371.486
63.016	3800.000	28376.999	320.016	4256.081
59.529	4000.000	28376.999	306.283	4146.714
56.199	4200.000	28376.999	293.016	4042.873
53.013	4400.000	28376.999	280.202	3944.111
49.958	4600.000	28376.999	267.825	3850.033
47.023	4800.000	28376.999	255.873	3760.291
44.201	5000.000	28376.999	244.334	3674.571
41.482	5200.000	28376.999	233.196	3592.591
38.859	5400.000	28376.999	222.446	3514.097
36.325	5600.000	28376.999	212.076	3438.856
33.876	5800.000	28376.999	202.073	3366.555
31.505	6000.000	28376.999	192.429	3297.290
29.207	6200.000	28376.999	183.134	3230.607

SPINTABLE MASS	0.0						
SPACE-TUG(EXPEN)	22625.0	2042.0	104.5	109.7	456.5	15000.0	28377.0
TE364-4(2300)	1043.0	88.0	9.	[illegible]	283.0	15[illegible]	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
166.196	800.000	27451.199	565.472	7594.433
151.894	[illegible]	27671.199	551.562	7137.572
140.479	1200.000	27891.199	538.456	6765.209
130.989	1400.000	28111.199	526.035	6450.319
122.870	1600.000	28331.199	514.217	6177.021
115.179	1800.000	28376.999	495.583	5927.479
108.141	2000.000	28376.999	475.752	5701.201
101.754	2200.000	28376.999	456.643	5495.519
95.901	2400.000	28376.999	436.215	5306.95
90.496	2600.000	28376.999	420.436	5132.870
85.470	2800.000	28376.999	403.276	4971.255
80.772	3000.000	28376.999	386.708	4820.505
76.359	3200.000	28376.999	370.707	4679.333
72.199	3400.000	28376.999	355.25[illegible]	4546.682
68.261	3600.000	28376.999	340.324	4421.676
64.524	3800.000	28376.999	325.901	4303.575
60.967	4000.000	28376.999	311.968	4191.749
57.573	4200.000	28376.999	298.508	4085.655
54.329	4400.000	28376.999	285.5[illegible]7	3984.82[illegible]
51.220	4600.000	28376.999	272.948	3888.824
48.236	4800.000	28376.999	260.821	3797.313
45.368	5000.000	28376.999	249.110	3709.947
42.607	5200.000	28376.999	237.895	3626.436
39.944	5400.000	28376.999	226.895	3546.513
37.374	5600.000	28376.999	216.367	3469.939
34.890	58[illegible]	28376.999	206.212	3396.491
32.487	6000.000	28376.999	196.419	3325.968
30.159	6200.000	28376.999	186.980	3258.183

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SPINTABLE MASS	0.0						
C-1S(PLANETARY)	13539.0	2219.0	61.0	18.0	439.8	30000.0	28377.0
APM-I ,	1710.0	144.0	9.0	0.0	297.0	15000.0	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
164.399	600.000	18310.000	67.975	8753.435
143.723	800.000	18530.000	63.414	8126.579
128.096	1000.000	18750.000	59.694	7637.522
115.650	1200.000	18970.000	56.543	7237.840
105.372	1400.000	19190.000	53.803	6900.596
96.660	1600.000	19410.000	51.377	6609.333
89.127	1800.000	19630.000	49.197	6353.296
82.511	2000.000	19850.000	47.219	6125.091
76.626	2200.000	20070.000	45.408	5919.426
71.339	2400.000	20290.000	43.739	5732.385
66.549	2600.000	20510.000	42.192	5560.993
62.177	2800.000	20730.000	40.751	5402.951
58.162	3000.000	20950.000	39.403	5256.414
54.456	3200.000	21170.000	38.139	5119.912
51.020	3400.000	21390.000	36.950	4992.239
47.819	3600.000	21610.000	35.828	4872.397
44.829	3800.000	21830.000	34.766	4759.547
42.025	4000.000	22050.000	33.761	4652.981
39.389	4200.000	22270.000	32.806	4552.095
36.904	4400.000	22490.000	31.898	4456.363
34.556	4600.000	22710.000	31.033	4365.347
32.333	4800.000	22930.000	30.218	4278.638
30.223	5000.000	23150.000	29.420	4195.894
28.218	5200.000	23370.000	28.667	4116.808
26.309	5400.000	23590.000	27.946	4041.108
24.489	5600.000	23810.000	27.255	3968.550
22.751	5800.000	24030.000	26.592	3898.917
21.088	6000.000	24250.000	25.956	3832.013
19.497	6200.000	24470.000	25.345	3767.660

SPINTABLE MASS	3.4						
D-1S(PLANETARY)	13539.0	2219.0	61.0	18.0	439.8	30000.0	28377.0
BII(2300)	1043.3	226.0	12.7	11.2	283.[illegible]	15[illegible]	[illegible]

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
188.470	200.000	17330.200	67.586	9503.062
159.363	400.000	17530.200	63.409	8649.081
138.942	600.000	17740.200	59.873	8024.514
123.279	800.000	17960.200	56.776	7529.910
111.079	1000.000	18180.200	54.100	7134.516
101.126	1200.000	18400.200	51.729	6814.997
92.743	1400.000	18620.200	49.592	6522.327
85.515	1600.000	18840.200	47.644	6274.669
79.173	1800.000	19060.200	45.852	6054.218
73.530	2000.000	19280.200	44.193	5855.535
68.456	2200.000	19500.200	42.649	5674.698
63.851	2400.000	19720.200	41.203	5508.779
59.641	2600.000	19940.200	39.848	5355.537
55.768	2800.000	20160.200	38.572	5213.214
52.186	3000.000	20380.200	37.368	5080.406
48.859	3200.000	20600.200	36.235	4955.973
45.756	3400.000	20820.200	35.151	4838.974
42.851	3600.000	21040.200	34.127	4728.624
40.124	3800.000	21260.200	33.153	4624.257
37.556	4000.000	21480.200	32.226	4525.310
35.132	4200.000	21700.200	31.342	4431.293
32.840	4400.000	21920.200	30.498	4341.784
30.666	4600.000	22140.200	29.691	4256.413
28.602	4800.000	22360.200	28.920	4174.855
26.638	5000.000	22580.200	28.181	4096.822
24.767	5200.000	22800.200	27.472	4022.357

SPINTABLE MASS	0.0						
D-1S(PLANETARY)	13539.0	2219.0	61.0	18.0	439.8	30000.0	28377.0
PM(2300)	1043.0	175.0	14.0	0.0	283.0	15000.0	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
199.290	200.000	17269.000	68.937	9810.356
166.116	400.000	17469.000	64.498	8850.671
143.708	600.000	17679.000	60.772	8172.120
126.895	800.000	17899.000	57.545	7645.21[illegible]
113.980	1000.000	18119.000	54.777	7229.259
103.543	1200.000	18339.000	52.337	6885.518
94.811	1400.000	18559.000	50.145	6592.429
87.320	1600.000	18779.000	48.152	6336.805
80.772	1800.000	18999.000	46.322	6110.044
74.965	2000.000	19219.000	44.630	5906.232
69.755	2200.000	19439.000	43.056	5721.134
65.036	2400.000	19659.000	41.586	5551.612
60.729	2600.000	19879.000	40.208	5395.278
56.774	2800.000	20099.000	38.912	5250.269
53.123	3000.000	20319.000	37.690	5115.103
49.729	3200.000	20539.000	36.534	4988.581
46.569	3400.000	20759.000	35.440	4869.716
43.615	3600.000	20979.000	34.402	4757.688
40.842	3800.000	21199.000	33.415	4651.805
38.234	4000.000	21419.000	32.475	4551.477
35.773	4200.000	21639.000	31.580	4456.209
33.447	4400.000	21859.000	30.725	4365.535
31.243	4600.000	22079.000	29.909	4279.094
29.150	4800.000	22299.000	29.128	4196.557
27.160	5000.000	22519.000	28.380	4117.612
25.265	5200.000	22739.000	27.664	4041.999

SPINTABLE MASS 0.0
D-1S(PLANETARY) 13539.0 2219.0 61.0 16.0 439.8 30000.0 28377.0
TE364-4(2300) 1043.0 88.0 9.0 [illegible] 283.[illegible] 15[illegible]

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
222.674	200.000	17177.000	71.196	10458.052
179.305	400.000	17377.000	66.416	9238.293
152.510	600.000	17587.000	62.309	8441.762
133.332	800.000	17807.000	58.825	7848.849
119.018	1000.000	18027.000	55.888	7392.722
107.665	1200.000	18247.000	53.373	7022.111
98.289	1400.000	18467.000	51.035	6709.811
90.323	1600.000	18687.000	48.964	6439.773
83.410	1800.000	18907.000	47.070	6201.786
77.314	2000.000	19127.000	45.322	5988.973
71.868	2200.000	19347.000	43.701	5796.483
66.955	2400.000	19567.000	42.189	5620.775
62.484	2600.000	19787.000	40.774	5459.179
58.388	2800.000	20007.000	39.445	5309.634
54.614	3000.000	20227.000	38.192	5170.512
51.117	3200.000	20447.000	37.010	5040.506
47.864	3400.000	20667.000	35.891	4918.548
44.827	3600.000	20887.000	34.830	4803.751
41.981	3800.000	21107.000	33.822	4695.375
39.306	4000.000	21327.000	32.863	4592.791
36.785	4200.000	21547.000	31.949	4495.459
34.404	4400.000	21767.000	31.078	4402.914
32.150	4600.000	21987.000	30.246	4314.753
30.012	4800.000	22207.000	29.450	4230.621
27.981	5000.000	22427.000	28.689	4150.205
26.047	5200.000	22647.000	27.960	4073.229

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SPINTABLE MASS	113.4						
SPACE-TUG(EXPEN)	22625.0	2755.4	104.5	109.7	456.5	15000.0	28377.0
APM-I	1710.0	257.4	9.0	0.0	297.0	15000.0	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
157.207	1000.000	28376.999	511.450	7309.069
144.038	1200.000	28376.999	490.363	6897.371
133.003	1400.000	28376.999	470.234	6548.218
123.509	1600.000	28376.999	450.956	6245.177
115.178	1800.000	28376.999	432.452	5977.572
107.753	2000.000	28376.999	414.662	5738.068
101.054	2200.000	28376.999	397.540	5521.419
94.948	2400.000	28376.999	381.045	5323.750
89.338	2600.000	28376.999	365.145	5142.118
84.146	2800.000	28376.999	349.810	4974.233
79.313	3000.000	28376.999	335.015	4818.282
74.793	3200.000	28376.999	320.738	4672.799
70.545	3400.000	28376.999	306.959	4536.584
66.538	3600.000	28376.999	293.658	4408.638
62.746	3800.000	28376.999	280.819	4288.124
59.146	4000.000	28376.999	268.426	4174.326
55.720	4200.000	28376.999	256.464	4066.633
52.451	4400.000	28376.999	244.920	3964.511
49.325	4600.000	28376.999	233.782	3867.497
46.330	4800.000	28376.999	223.036	3775.179
43.456	5000.000	28376.999	212.671	3687.193
40.693	5200.000	28376.999	202.677	3603.220
38.032	5400.000	28376.999	193.043	3522.960
35.467	5600.000	28376.999	183.759	3446.152
32.990	5800.000	28376.999	174.815	3372.555
30.596	6000.000	28376.999	166.202	3301.952
28.279	6200.000	28376.999	157.912	3234.138

SPINTABLE MASS	113.4					
SPACE-TUG(EXPEN)	22625.0	2755.4	104.5	109.7	456.5 1[illegible]	28377.0
TE364-4(2300)	1043.0	201.4	9.0	0.0	283.0 15000.0	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
163.417	800.000	27564.999	558.757	7510.990
149.331	1000.000	27784.999	545.141	7052.671
138.1[illegible]	1200.000	28004.999	532.309	6690.453
128.770	1400.000	28224.999	52[illegible].144	6379.363
120.557	1600.000	28376.999	505.690	6106.510
112.840	1800.000	28376.999	485.461	5858.417
105.922	2000.000	28376.999	465.980	5635.443
99.644	2200.000	28376.999	447.2[illegible]	5432.841
93.891	2400.000	28376.999	429.100	5247.150
88.575	2600.000	28376.999	411.629	5073.765
83.633	2800.000	28376.999	394.766	4916.677
79.011	3000.000	28376.999	378.483	4768.392
74.669	3200.000	28376.999	362.759	4629.364
70.574	3400.000	28376.999	347.570	4498.818
66.698	3600.000	28376.999	332.898	4375.798
63.018	3800.000	28376.999	318.725	4259.571
59.514	4000.000	28376.999	305.033	4149.517
56.170	4200.000	28376.999	291.806	4045.[illegible]99
52.972	4400.000	28376.999	279.031	3945.850
49.907	4600.000	28376.999	266.693	3851.35[illegible]
46.964	4800.000	28376.999	254.778	3761.267
44.135	5000.000	28376.999	243.276	3675.249
41.41[illegible]	5200.000	28376.999	232.173	3593.014
38.783	5400.000	28376.999	221.458	3514.372
36.245	5600.000	28376.999	211.122	3438.873
33.793	5800.000	28376.999	201.153	3366.511
31.419	6000.000	28376.999	191.541	3297.017

SPINTABLE MASS	113.4						
D-1S(PLANETARY)	13539.0	2332.4	61.0	18.0	439.8	30[illegible]	28377.0
APM-I	1710.0	257.4	9.0	0.0	297.0	15[illegible]	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
161.899	600.000	18423.400	67.067	8681.575
141.447	800.000	18643.400	62.563	8058.763
126.0[illegible]6	1000.000	18863.400	58.893	7573.399
113.716	1200.000	19083.400	55.787	7177.083
103.575	1400.000	19303.400	53.089	6842.934
94.983	1600.000	19523.400	50.700	6554.512
87.555	1800.000	19743.400	48.556	6301.097
81.033	2000.000	19963.400	46.61[illegible]	6075.317
75.235	2200.000	20183.400	44.829	5671.899
70.025	2400.000	20403.400	43.187	5686.94[illegible]
65.305	2600.000	20623.400	41.666	5517.504
60.998	2800.000	20843.400	40.249	5361.271
57.042	3000.000	21063.400	38.924	5216.431
53.391	3200.000	21283.400	37.681	5081.513
50.004	3400.000	21503.400	36.511	4955.33[illegible]
46.850	3600.000	21723.400	35.407	4836.894
43.903	3800.000	21943.400	34.363	4725.362
41.140	4000.000	22163.400	33.374	4620.039
38.541	4200.000	22383.400	32.435	452[illegible].325
36.091	4400.000	22603.400	31.541	4425.7[illegible]
33.776	4600.000	22823.400	30.690	4335.733
31.584	4800.000	23043.400	29.878	4250.017
29.503	5000.000	23263.400	29.1[illegible]3	4168.214
27.525	5200.000	23483.400	26.361	4090.[illegible]21
25.642	5400.000	23703.400	27.651	4015.179
23.846	5600.000	23923.400	26.971	3943.421

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SPINTABLE MASS	113.4						
D-1S(PLANETARY)	13539.6	2332.4	61.0	18.6	439.8	3[illegible]	28377.0
TE364-4(2300)	1043.0	201.4	9.0	0.0	283.0	15000.0	

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
218.848	200.000	17250.400	69.946	1[illegible]356.6[illegible]7
175.951	400.000	17490.400	65.251	9143.299
149.510	600.000	17700.400	61.221	8352.83[illegible]
13[illegible].621	800.000	17920.400	57.812	7765.602
116.542	1000.000	18140.400	54.938	7314.579
105.387	1200.000	18360.400	52.431	6948.571
96.183	1400.000	18580.400	5[illegible].196	664[illegible].442
88.366	1600.000	18800.400	48.173	6374.198
81.585	1800.000	19020.400	46.322	6139.677
75.606	2000.000	19240.400	44.616	5930.035
70.265	2200.000	19460.400	43.[illegible]32	[illegible]74[illegible].467
65.447	2400.000	19680.400	41.555	5567.44[illegible]
61.062	2600.000	19900.400	40.171	54[illegible]8.339
57.045	2800.000	20120.400	38.872	5261.097
53.341	3000.000	20340.400	37.647	5124.115
49.91[illegible]	3200.000	20560.400	36.490	4996.101
46.718	3400.000	20780.400	35.395	4876.001
43.736	3600.000	21000.400	34.356	4762.941
40.941	3800.000	21220.400	33.369	4656.19[illegible]
38.314	4000.000	21440.400	32.429	4555.130
35.838	4200.000	21660.400	31.534	4459.230
33.498	4400.000	21880.400	30.680	4363.033
31.283	4600.000	22100.400	29.865	4281.142
29.181	4800.000	22320.400	29.084	4198.207
27.183	5000.000	22540.400	28.337	4118.923
25.280	5200.000	22760.400	27.622	4043.017
23.466	5400.000	22980.400	26.935	397[illegible].248

SPINTAELE MASS C.C
SPACE-TUG(EXPEN) 22625.C 2642.0 27.0 1C9.7 456.5 15000.0 28377.0

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
162.303	200.000	25603.699	698.847	7435.528
154.046	400.000	25803.699	680.002	7177.092
146.126	600.000	26013.699	661.394	6925.897
138.537	800.000	26233.699	643.056	6682.096
131.575	1000.000	26453.699	625.775	6455.688
125.159	1200.000	26673.699	609.445	6244.642
119.220	1400.000	26893.699	593.975	6047.257
113.704	1600.000	27113.699	579.287	5862.097
108.563	1800.000	27333.699	565.314	5687.937
103.757	2000.000	27553.699	551.995	5523.723
99.251	2200.000	27773.699	539.279	5368.544
95.017	2400.000	27993.699	527.121	5221.606
91.027	2600.000	28213.699	515.478	5082.21[illegible]
87.072	2800.000	28376.999	501.894	4947.376
82.783	3000.000	28376.999	481.957	4812.499
78.693	3200.000	28376.999	462.720	4684.127
74.786	3400.000	28376.999	444.150	4561.811
71.048	3600.000	28376.999	426.218	4445.146
67.467	3800.000	28376.999	408.898	4333.762
64.031	4000.000	28376.999	392.166	4227.327
60.729	4200.000	28376.999	376.000	4125.532
57.553	4400.000	28376.999	360.379	4028.096
54.494	4600.000	28376.999	345.283	3934.765
51.544	4800.000	28376.999	330.695	3845.295
48.697	5000.000	28376.999	316.597	3759.465
45.946	5200.000	28376.999	302.975	3677.073
43.285	5400.000	28376.999	289.813	3597.924
40.716	5600.000	28376.999	277.097	3521.839
38.214	5800.000	28376.999	264.815	3448.631
35.795	6000.000	28376.999	252.953	3378.201

SPINTABLE MASS C.0
D-1S(PLANETARY) 13539.0 2219.0 27.0 13.0 439.8 3[illegible] 28377.0

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
128.442	200.000	16003.000	74.233	7736.133
119.608	400.000	16203.000	70.872	7456.[illegible]05
111.282	600.000	16413.000	67.643	7187.529
103.434	800.000	16633.000	64.544	6930.297
96.343	1000.000	16853.000	61.698	6694.265
89.897	1200.000	17073.000	59. 74	6476.583
84.006	1400.000	17293.000	56.644	6274.941
78.596	1600.000	17513.000	54.388	6087.429
73.609	1800.000	17733.000	52.286	5912.450
68.993	2000.000	17953.000	50.322	5748.658
64.703	2200.000	18173.000	48.483	5594.903
60.711	2400.000	18393.000	46.758	5459.199
56.980	2600.000	18613.000	45.135	5313.696
53.485	2800.000	18833.000	43.606	5184.651
50.203	3000.000	19053.000	42.163	5062.416
47.115	3200.000	19273.000	40.798	4946.420
44.203	3400.000	19493.000	39.506	4836.157
41.451	3600.000	19713.000	38.281	4731.178
38.848	3800.000	19933.000	37.118	4631.084
36.379	4000.000	20153.000	36.012	4535.516
34.035	4200.000	20373.000	34.959	4444.151
31.807	4400.000	20593.000	33.957	4356.699
29.684	4600.000	20813.000	33.000	4272.895
27.661	4800.000	21033.000	32.086	4192.503
25.729	5000.000	21253.000	31.213	4115.304
23.883	5200.000	21473.000	30.378	4041.1[illegible]
22.117	5400.000	21693.000	29.579	3969.710
20.425	5600.000	21913.000	28.813	3900.966
18.803	5800.000	22133.000	28.078	3834.716

SPINTABLE MASS 113.4
SPACE-TUG(EXPEN) 22625.0 2755.4 27.0 109.7 456.5 15000.0 28377.0

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
157.530	200.000	25717.099	688.020	7286.536
149.681	400.000	25917.099	669.812	7039.056
142.130	600.000	26127.099	651.802	6797.916
134.876	800.000	26347.099	634.023	6563.358
128.204	1000.000	26567.099	617.245	6345.499
122.042	1200.000	26787.099	601.369	6141.289
116.328	1400.000	27007.099	586.311	595[illegible].371
111.[illegible]1[illegible]	1600.000	27227.099	571.999	5771.02[illegible]
106.046	1800.000	27447.099	558.370	5602.111
101.399	2000.000	27667.099	545.368	5442.660
97.036	2200.000	27887.099	532.945	5291.821
92.931	2400.000	28107.099	521.057	5148.852
89.060	2600.000	28327.099	509.667	5013.100
84.835	2800.000	28376.999	491.528	4877.011
80.651	3000.000	28376.999	471.956	4745.546
76.657	3200.000	28376.999	453.066	4620.349
72.839	3400.000	28376.999	434.829	4500.994
69.184	3600.000	28376.999	417.216	4387.095
65.678	3800.000	28376.999	40[illegible].2[illegible]2	4278.302
62.313	4000.000	28376.999	383.764	4174.293
59.077	4200.000	28376.999	367.881	4074.781
55.962	4400.000	28376.999	352.533	3979.492
52.960	4600.000	28376.999	337.701	3888.179
50.[illegible]64	4800.000	28376.999	323.368	3800.613
47.267	5000.000	28376.999	309.517	3716.580
44.564	5200.000	28376.999	296.134	3635.882
41.947	5400.000	28376.999	283.2[illegible]4	3558.334
39.414	5600.000	28376.999	270.713	3485.763
36.958	5800.000	28376.999	258.649	3412.003
34.576	6000.000	28376.999	246.999	3342.907

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OF POOR QUALITY

SPINTABLE MASS 113.4
D-1S(PLANETARY) 13539.0 2332.4 27.0 18.0 439.8 300[illegible] 28377.0

C3(KM/SEC**2)	NET PAYLOAD(KG)	INI. MASS(KG)	GRAV. LOSS(MPS)	TOTAL DV(MPS)
123.316	200.000	16116.400	72.291	7574.147
115.002	400.000	16316.400	69.093	7308.011
107.134	600.000	16526.400	66.012	7052.083
99.692	800.000	16746.400	63.048	6806.183
92.946	1000.000	16966.400	60.319	6579.924
86.796	1200.000	17186.400	57.798	6370.771
81.161	1400.000	17406.400	55.461	6176.629
75.976	1600.000	17626.400	53.286	5995.763
71.186	1800.000	17846.400	51.257	5826.703
66.744	2000.000	18066.400	49.359	5668.216
62.612	2200.000	18286.400	47.580	5519.241
58.757	2400.000	18506.400	45.999	5378.853
55.150	2600.000	18726.400	44.336	5246.291
51.768	2800.000	18946.400	42.852	512[illegible].832
48.588	3000.000	19166.400	41.450	5001.880
45.593	3200.000	19386.400	40.124	4888.893
42.765	3400.000	19606.400	38.867	4781.411
40.092	3600.000	19826.400	37.674	4678.993
37.559	3800.000	20046.400	36.541	4581.279
35.156	4000.000	20266.400	35.463	4487.916
32.873	4200.000	20486.400	34.437	4398.602
30.700	4400.000	20706.400	33.458	4313.062
28.631	4600.000	20926.400	32.524	4231.045
26.654	4800.000	21146.400	31.632	4152.325
24.768	5000.000	21366.400	30.778	4076.693
22.963	5200.000	21586.400	29.962	4003.961
21.236	5400.000	21806.400	29.180	3933.955
19.581	5600.000	22026.400	28.430	3860.515

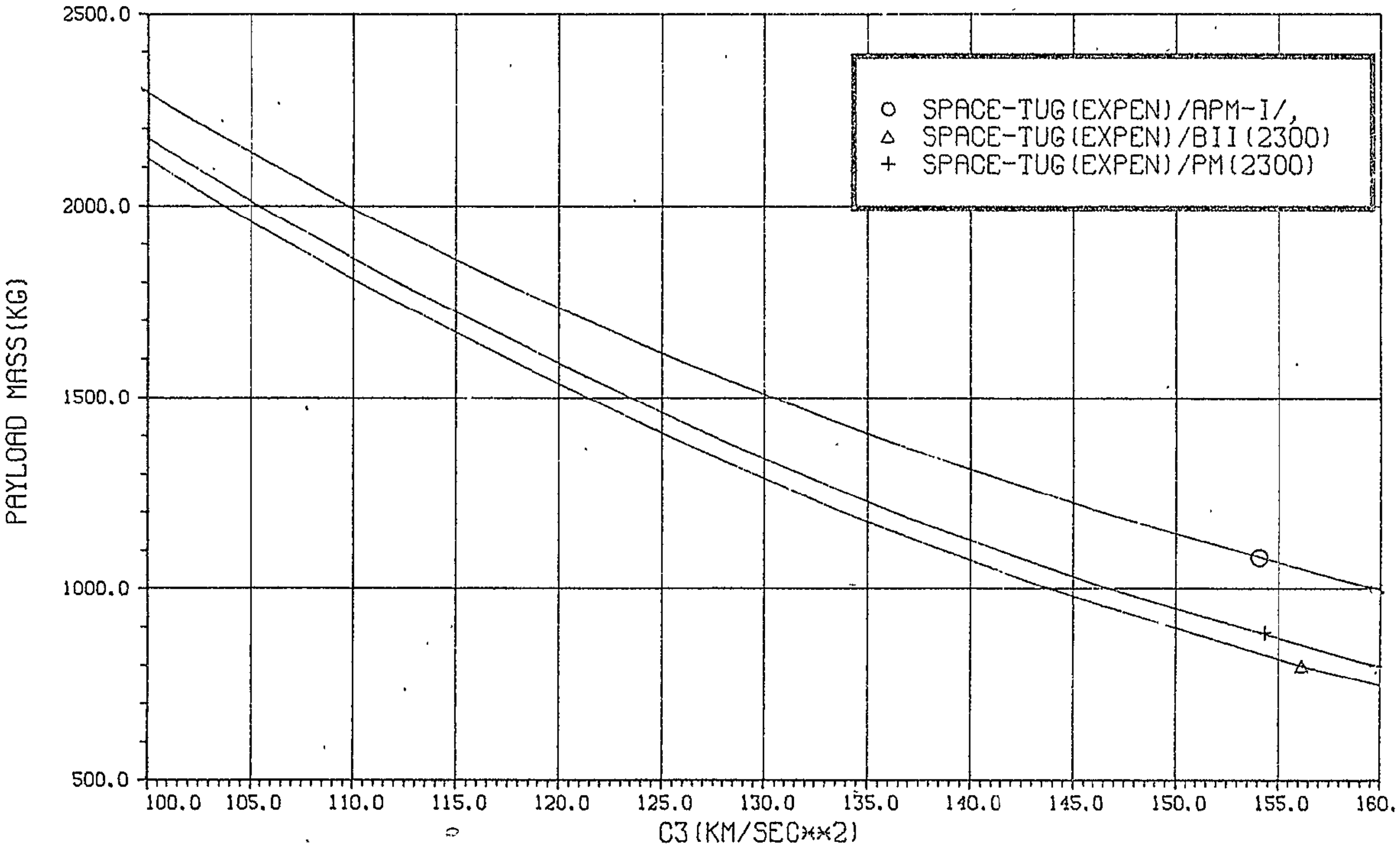

FIGURE D-1. SPACE-TUG/3-AXIS STAB. KICK STGS.

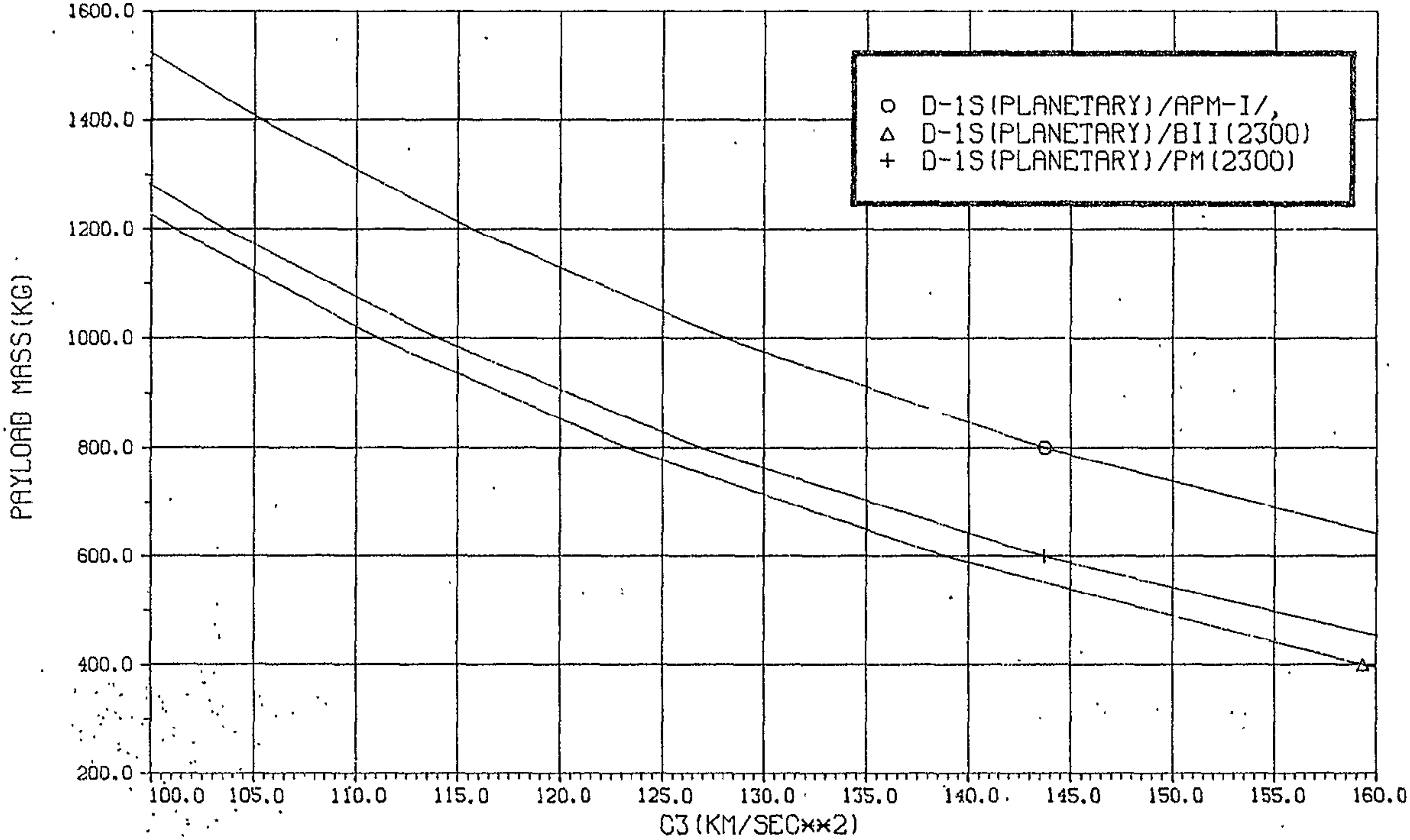

FIGURE D-2. CENTAUR D-1S/3-AXIS STAB. KICK STGS.

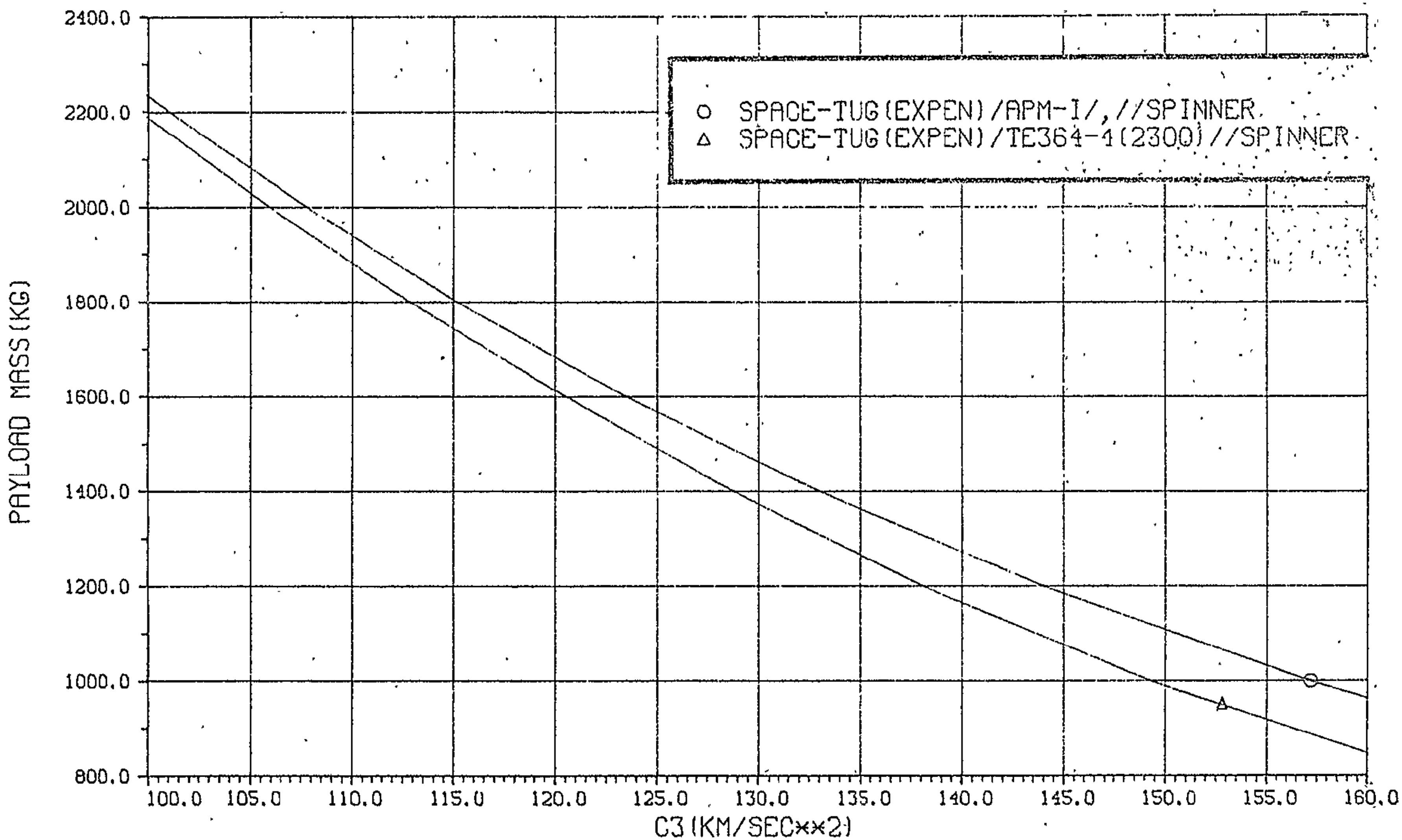

FIGURE D-3. SPACE-TUG/SPIN STAB. KICK STGS.

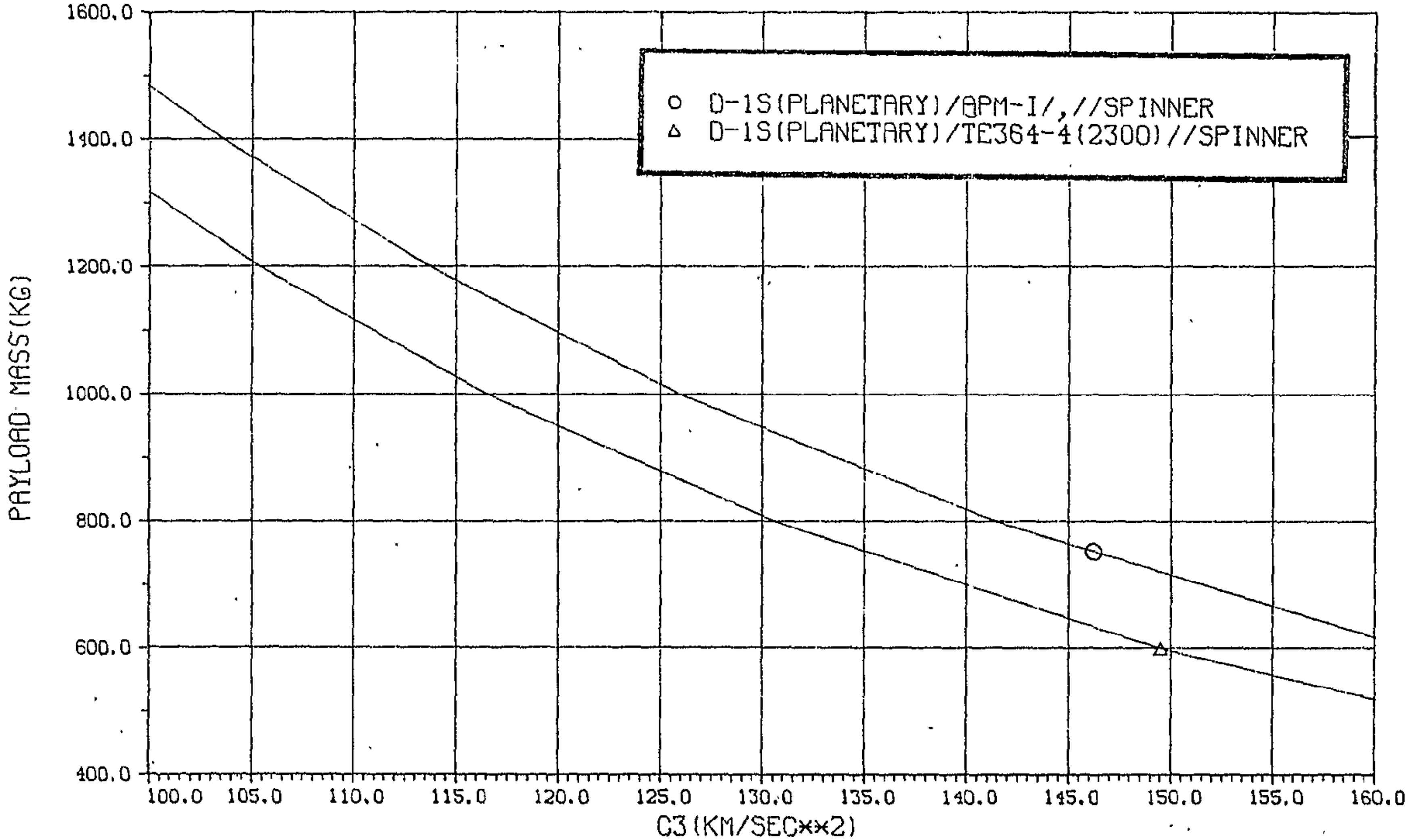

FIGURE D-4. CENTAUR D-1S/SPIN STAB. KICK STGS.

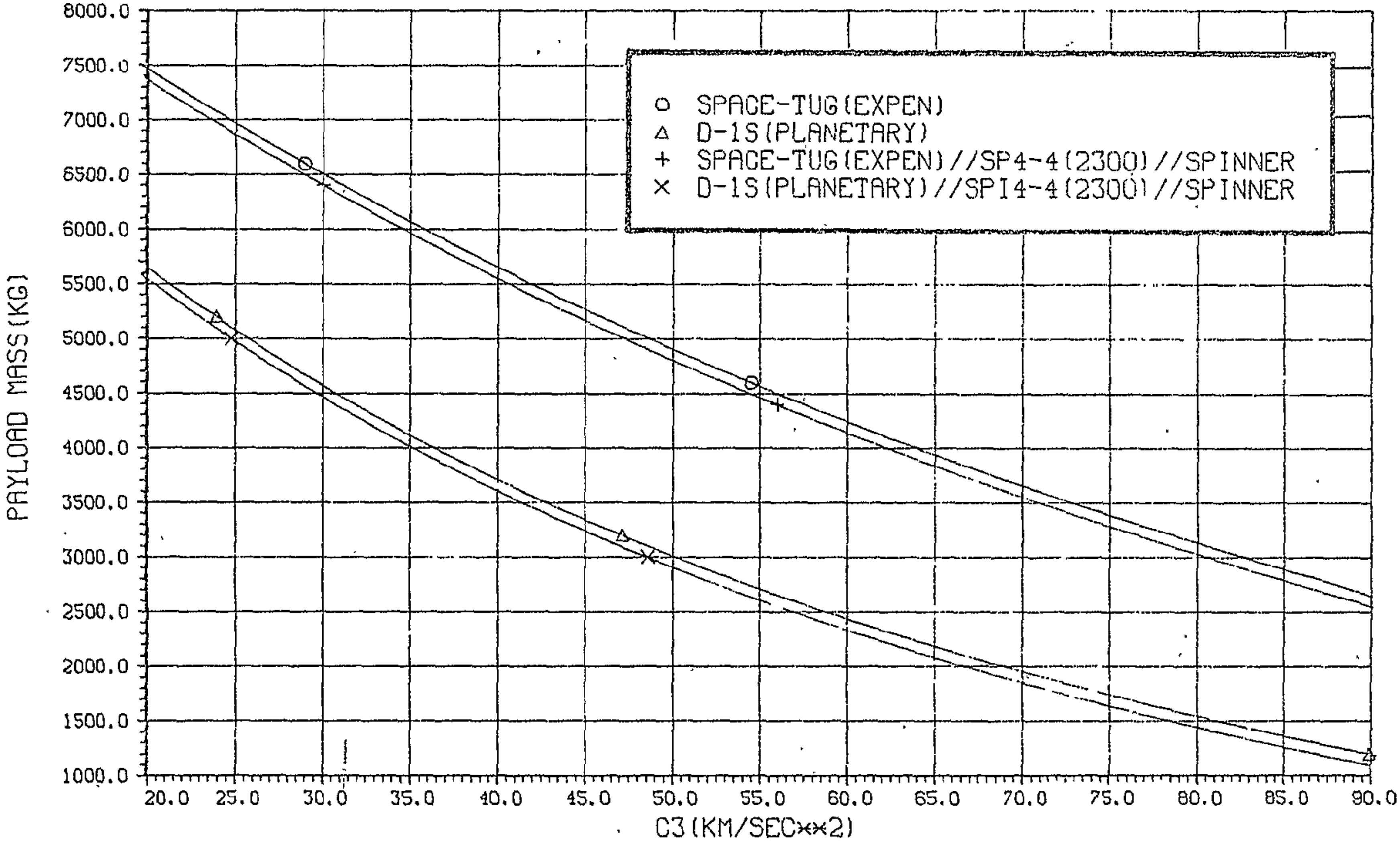

FIGURE D-5. UNAIDED UPPER STAGES

c) Only the principal term of the earth's gravitational potential (i.e., μ/r) is included in the burn simulation ($\mu = 1.4076468 \times 10^{16}$ ft^3/sec^2). The resulting equations of motion are integrated numerically.

d) The thrust vector is always aligned with the current inertial velocity vector of the vehicle.

e) The total vehicle mass at upper stage ignition is required to be less than or equal to a specified upper limit, W_{UB} (see table below). This corresponds to the nominal Shuttle payload capability in a 160 km circular orbit (65,000 lb_m). The combined mass of the vehicle and adapter-pallet must not exceed 65,000 lb_f. The values of W_{UB} are a function of the upper stage only.

Upper Limit of Vehicle Mass
at Upper Stage Ignition

Upper Stage	W_{UB} (kg)
Space-Tug	28,622
D-1S Centaur	26,137

f) If the mass of the fully loaded vehicle would exceed W_{UB} the upper stage fuel is off-loaded until the total vehicle mass equals W_{UB}. The kick stage is never off-loaded.

g) Each stage is burned until its fuel is depleted.

h) After a stage has burned out it is jettisoned. The jettisoned mass includes the burnout mass plus the interstage adapter.

i) The time interval between upper stage burnout and kick stage ignition is assumed to be zero.

j) The burn simulation algorithm permits specification of the following parameters for the upper stages and kick stages. (Values used in this study are given in Table D-1.)

- Upper stage burnout mass
- Upper stage usable propellant mass
- Upper stage nonimpulsive inert mass. This mass consists of the propellant and other fluids that are present at first burn ignition but consumed at upper stage burnout. They do not contribute to the vehicle thrust. This mass is assumed to be expended at uniform rate from a specified value at first burn ignition to zero at upper stage burnout.

Table D-1. Propulsion and Mass Characteristics of Stage Vehicles Used in Performance Evaluation

Stage	Data Source	Burnout Mass kg (lb_m)	Usable Propellant Mass kg (lb_m)	Nonimpulsive Inerts Mass kg (lb_m)	Interstage Adapter kg (lb_m)	Specific Impulse (sec)	Thrust Magnitude (lb_f)
D-1S Centaur (Planetary)	Centaur/Shuttle Integration Study Final Report (Vol. II) Contract NAS3-16786	2219 (4892)	13539 (29854)	18 (40)	61 (135)	439.8	30000
Space-Tug (Expendable)	Baseline Space-Tug Configuration Definition, MSFC 68M00039-2, MSFC Science and Eng. Dir. MSFC-EA-EP01, 15 July 1974, pp 41-42, pp 79 and 25	2642 (5825)	22625 (49889)	109 (241)	104 (230)	456.5	15000
Burner II (2300	Report No. BMI-NLVP-TM-73-4 on Space Shuttle Expen. Upper Stages to NASA, Contract No. NASw-2018, 28 Dec. 1973, pp B-4	226 (498.2)	1043 (2300)	11.2 (24.8)	12.7 (28)	283	15000
TE 364-4 (2300)	(Pioneer F version) R. Hofstetter, Pioneer Launch Vehicle and Operations, Mar. 1973	88 (194)	1043 (2300)	0 0	9 19.8	283	1500
APM-1 Formerly designated SPM (1800)	T. W. Behm, JPL (informal communication)	144 (318)	1710 (3771)	0 0	9 19.8	297	15000
PM (2300)	D. Dugan, NASA Ames (informal communication)	175 (386)	1043 (2300)	0 0	14 31	283	1500

- Upper stage/kick stage adapter mass
- Kick stage burnout mass
- Kick stage usable propellant mass
- Kick stage nonimpulsive inert mass
- Kick stage/payload adapter mass. This mass equals a specified constant plus the term MAX [0, 0.10 (payload mass - 500 kg)].

k) The net payload determined by the simulation consists of everything above the kick stage adapter.

l) If the mission requires a spin-stabilized kick stage the spin table mass is assumed to be 113.34 kg. In the simulation the spin table mass is added to the upper stage burnout mass.

APPENDIX E

OPTIMIZATION OF PLANETARY INSERTION MANEUVERS

An automatic search routine is described which is designed to determine planetary insertion maneuvers with minimum propellant requirements. Maneuver constraints such as fixed thrust orientation or constant rate of change of thrust orientation can be imposed readily on the search routine.

Assumptions and constraints in defining the optimization approach and the algorithm used in the study are described below.

For purposes of illustration a vehicle with two propulsion modules* operating in tandem (i.e., the Mercury orbiter) is assumed. Generalization to other configurations can be made without difficulty.

1. ASSUMPTIONS AND CONSTRAINTS

1) The vehicle being inserted into planetary orbit consists of a payload of mass m_P and two similar stages. Each stage is required to have the same fuel capacity, m_C. The inert mass is $c_{11} + c_{21}\, m_C$ for the first stage and $c_{12} + c_{22}\, m_C$ for the second stage. There is an interstage adapter of mass m_A and a payload adapter of mass m_{PA}. The quantities m_P, c_{11}, c_{21}, c_{12}, c_{22}, m_A, m_{PA} are all specified constants.

2) The thrust and specific impulse of the first stage, F_1, I_1, and of the second stage, F_2, I_2, are specified constants.

3) The thrust vector is required to be coplanar with the plane of the planetary approach hyperbola.

4) The in-plane thrust direction must be specified although it is unrestricted.

5) The magnitude of the incoming V-infinity vector, V_∞, is specified. The periapse radius of the incoming hyperbola, R_p, is unspecified. R_p will be determined by the algorithm.

---

*The propulsion module will be referred to as "stage" in the discussion that follows.

6) The periapse radius and total energy of the target orbit, R_T and E_T, are specified.

7) The time of first burn ignition, T, is specified. T = 0 at periapse of the approach hyperbola.

8) The coast time between burns is zero. This condition could be relaxed without difficulty.

9) A preinsertion propellant budget (for midcourse corrections, etc.), m_{PI}, and a post-insertion propellant reserve, m_R, are specified constants. $m_{PI} \le m_C$ and $m_R \le m_C$.

2. PROBLEM DEFINITION

Given m_P, c_{11}, c_{21}, c_{12}, c_{22}, m_A, m_{PA}, F_1, I_1, F_2, I_2, the in-plane thrust direction, a V_∞, R_T, E_T and T, the algorithm described below shall determine R_p and the smallest value of m_C which results in attainment of the specified targets R_T and E_T. Initially, all propellant tanks are full, and at burnout of the insertion maneuver only the propellant reserve, m_R, remains.

3. ALGORITHM

1) Set $R_p = R_T$.

2) Obtain an initial guess for m_C. This number may either be externally supplied or computed assuming ideal thrust maneuvers.

3) Compute the vehicle state vector at periapse of the approach hyperbola assuming no insertion burn occurs

$$S_0(1) = R_p$$

$$S_0(2) = 0$$

$$S_0(3) = 0$$

$$S_0(4) = 0$$

$$S_0(5) = \sqrt{2\,E_T + 2\mu/R_p}$$

$$S_0(6) = 0$$

The coordinate system has its x-axis along the line of apsides of the incoming hyperbola and the z-axis along the angular momentum vector of the incoming hyperbola. The time is zero at periapse on the incoming hyperbola assuming no insertion burn.

4) Propagate* the periapse state (i.e., $\overline{S}_0$) backwards to T. Call this state vector $\overline{S}_1$.

5) Compute the first-stage burn time, $t_{B1} = (m_C - m_{PI}) I_1/F_1$

- The mass of the fully loaded vehicle, m_{FL}, is given by

$$m_{FL} = m_P + m_C + c_{11} + c_{12} m_C + m_C + c_{12}$$
$$+ c_{12} m_C + m_A + m_{PA}$$

- The vehicle mass at the beginning of the first burn is $m_{FL} - m_{PI}$.
- The propellant used during the first insertion burn is given by: $m_C - m_{PI}$.

6) Propagate $\overline{S}_1$ to $T + t_{B1}$. Call the new state $\overline{S}_2$.

7) Compute the second-stage burn time, $t_{B2} = (m_C - m_R) I_2/F_2$

- The vehicle mass at the beginning of the second insertion burn is: $m_P + c_{12} + c_{22} m_C + m_C + m_{PA}$
- The propellant used during the second insertion burn is: $m_C - m_R$
- The ignition time of the second insertion burn is the same as the burnout time of the first insertion maneuver (i.e., $T + t_{B1}$).

8) Propagate $\overline{S}_2$ to $T + t_{B1} + t_{B2}$. Call the new state $\overline{S}_3$.

9) Compute the periapse radius and total energy corresponding to S_3. Call these variables r_T and e_T respectively.

10) If $|r_T - R_T|$ and $|e_T - E_T|$ are less than specified tolerances the problem is solved (i.e., the current values of R_p and m_C define the insertion trajectory that meets the given targets). If the tolerances are not met, continue.

*This means update the state vector by numerical integration or any other means. The fidelity of the simulation is constrained only by the conditions explicitly called out above. Note that the gravitational model is unconstrained but the thrust, specific impulse and thrust direction are.

11) Compute a new estimate of R_p. A simple offset method works very well. More specifically, the new estimate of R_p is given by the formula: $R_p + (r_T - R_T)$.

12) Compute a new estimate of m_C (see details in the next section).

13) Return to step 3).

4. m_C UPDATE PROCEDURE

On the very first iteration m_C is determined by step 2 above. For the second and third passes m_C in incremented by a constant. For the fourth and subsequent iterations the following procedure is used.

Let w_i, x_i, y_i, and z_i denote: stage propellant capacity, the approach hyperbola periapse radius, the periapse radius at insertion maneuver burnout, and the total energy at insertion maneuver burnout on the i^{th} iteration. The physical problem is such that when w_i and x_i are given, y_i and z_i are computed by the above algorithm. The problem considered here is that of determining w_{i+1} and x_{i+1} such that: $Y_{i+1} = R_T$ and $z_{i+1} = E_T$. Closed form solutions for these quantities do not exist; at best, a convergent sequence may be calculated. As noted previously

$$x_{i+1} = x_i + (y_i - R_T)$$

is used here as an estimate for hyperbolic radius.

Clearly,

$$z_{i+1} = f(w_{i+1}, x_{i+1}); \text{ f is unknown}$$

or, equivalently,

$$z_{i+1} = f(w_i + \Delta w, x_i + \Delta x)$$

Now, assuming Δw and Δx are small it follows that

$$z_{i+1} = f(w_i, x_i) + \Delta w \frac{\partial f}{\partial w} + \Delta x \frac{\partial f}{\partial x} + \cdots + \text{H.O.T.}$$

In the region near w_i and x_i it may be further assumed that the partial derivatives of f are constants. This leads to the relation

$$z_{i+1} - z_i = (w_{i+1} - w_i) A + (x_{i+1} - x_i) B$$

or

$$z_{i+1} - z_i = (w_{i+1} - w_i) A + (y_i - R_T) B \tag{1}$$

where A and B are constants. Now, since i in the above equation may be any integer it follows that

and

$$\left.\begin{aligned} z_i - z_{i-1} &= (w_i - w_{i-1}) A + (y_{i-1} - R_T) B \\ z_{i-1} - z_{i-2} &= w_{i-1} - w_{i-2}) A + (y_{i-1} - R_T) B \end{aligned}\right\} i > 2$$

From these two equations A and B may be computed then substituted into equation (1) and w_{i+1} may be computed (for this calculation $z_{i+1} = E_T$).

5. MINIMUM PROPELLANT PLANETARY INSERTION

The above algorithm determines the minimum propellant mass required for insertion into a spectified orbit when T and C are given where C denotes the set of constants: m_P, c_{11}, c_{21}, c_{12}, c_{22}, m_A, m_{PA}, F_1, I_1, F_2, I_2, V_∞, R_T, E_T. To find the value of T that yields the overall minimum propellant mass a one-dimensional optimization problem, requiring repeated applications of the algorithm, must be solved. The computer program implementing this approach employs the above algorithm and a "golden section optimization routine" to determine the absolute minimum propellant mass when the set of constants, C, is given.

APPENDIX F

SUPPORTING DATA ON ORBIT INSERTION PERFORMANCE

1. MERCURY ORBIT INSERTION WITH FIXED AND VARIABLE THRUST ORIENTATION

Optimum and near-optimum orbit insertion modes at Mercury were determined by a systematic performance optimization technique (see Appendix E) for given arrival conditions and a specified periapsis altitude (500 km), periapsis location and eccentricity of the capture orbit. Results were summarized in Section 7 of Volume II. Table F-1 lists maneuver requirements for tandem and single-stage Mercury orbit insertion, for earth- and space-storable propellants, and for fixed and variable thrust orientations. The maneuver requirements correspond to mission option 1 (see Section 2, Volume II) and propellant mass characteristics reflect the initial inert weight assumptions stated in that section. Although these results do not represent the final performance characteristics given in Section 7, they are useful in illustrating the relatively minor performance differences between the optimum fixed thrust pointing mode and the variable thrust pointing mode, where the thrust vector is oriented parallel and opposite to the velocity vector.

Comparison of the single-stage and tandem-stage orbit insertion modes shows the very large increase in propellant mass and total spacecraft mass if the inefficient single-stage insertion procedure were to be used. This would make the use of the Mercury mission module for outer-planet orbit missions quite impractical.

Figure F-1 illustrates the sensitivity of initial spacecraft mass and propellant requirements to thrust initiation time for both variable and fixed thrust orientations. It also shows the comparatively small difference between the two thrust pointing modes.

Mariner class spacecraft can implement a variable thrust pointing maneuver quite readily, using a stored program of orientation commands and an attitude gyro. Pioneer class spacecraft preferably maintain a fixed attitude during the maneuver. The results presented above show

Table F-1. Mercury Orbit Insertion Performance Characteristics and Propulsion Module Sizing Data

I_{sp} (sec)	No. of Stages Used	Thrust Orientation Mode	Maneuver Timing: Thrust Initiation Time[1] (sec)	Maneuver Timing: Burn Time[2] (sec)	Approach Hyperbola Periapsis Altitude (km)	Periapsis Angle Shift[3] (deg)	Weight Characteristics, kg (lb_m): Flight Spacecraft Initial Mass	Weight Characteristics, kg (lb_m): Stage Inert Mass[2]	Weight Characteristics, kg (lb_m): Propellant Mass[2]
Module A, Payload Mass 340 kg									
376	2	Variable	-734	557	604	21.0	1291 (2847)	71 (157)	404 (891)
		Fixed	-734	561	483	22.0	1297 (2860)	72 (159)	407 (897)
296	2	Variable	-1059	767	649	25.0	2002 (4414)	125 (276)	706 (1557)
		Fixed	-1068	778	458	26.0	2028 (4472)	126 (278)	717 (1581)
376	1	Variable	-889	1350	640	25.0	1492 (3290)	172 (379)	979 (2159)
		Fixed	-892	1365	469	26.0	1505 (3319)	175 (386)	990 (2183)
296	1	Variable	-1824	2457	790	18.0	3003 (6622)	400 (882)	2264 (4992)
		Fixed	-1752	2468	361	30.0	3015 (6648)	401 (884)	2274 (5014)
Module B, Payload Mass 550 kg									
376	2	Variable	-1278	939	699	29.6	2152 (4745)	120 (265)	681 (1502)
		Fixed	-1295	953	426	30.4	2177 (4800)	122 (269)	692 (1526)
296	2	Variable	-1931	1327	772	28.1	3426 (7510)	216 (476)	1222 (2695)
		Fixed	-1946	1357	332	32.3	3492 (7700)	221 (487)	1250 (2756)
376	1	Variable	-1561	2318	763	36.0	2528 (5574)	296 (653)	1681 (3707)
		Fixed	-1624	2383	369	34.0	2583 (5096)	305 (673)	1728 (3810)
296[3]	1	Variable[4]	-4211	5492	1308	6	6503[5] (14339)	893 (1969)	5060 (11156)

Assumptions:
Thrust level 600 lb_f (2730 N)
Mission Type 1 (launch date 19 June 1988)
Midcourse and orbit trim maneuvers not included
Preliminary inert weight scaling laws:
$W_i = 0.163\ W_p + 18.1$ kg (40 lb_m)
Mercury orbit: periapsis altitude 500 km; e = 0.8

Legend:
[1] Relative to periapsis passage of approach hyperbola
[2] Each stage
[3] Angle between apsidal line of incoming hyperbola and elliptical orbit
[4] Maneuver not feasible with fixed thrust orientation in this case
[5] Gross mass exceeds Shuttle/Space Tug capability

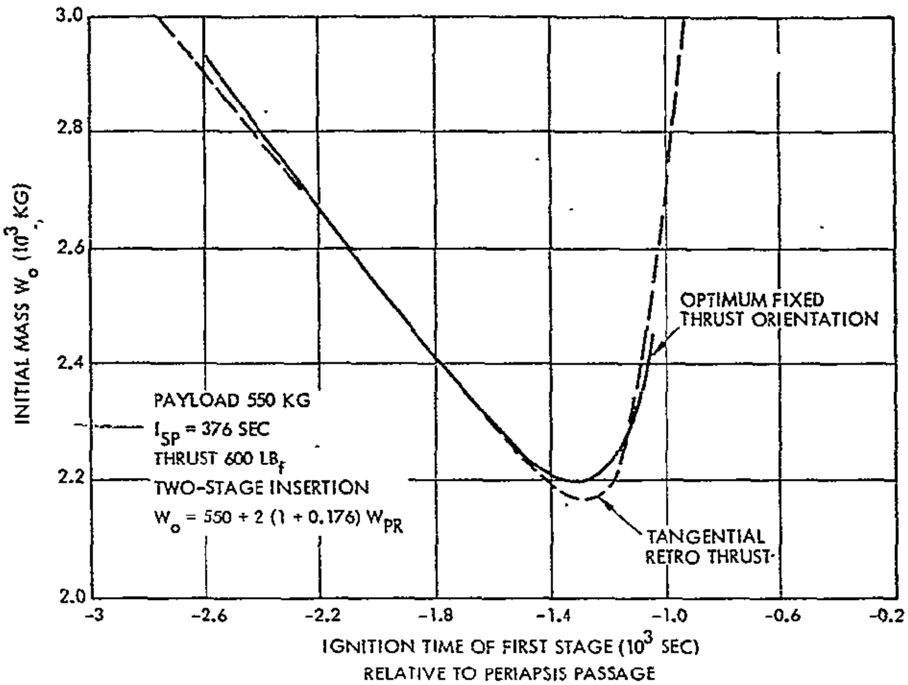

Figure F-2. Performance Comparison of Two Retro-Thrust Modes Versus Ignition Time for Mercury Orbiter

that the greater simplicity of a fixed maneuver attitude in the case of Pioneer class payload outweighs the performance gain obtainable by introducing the more sophisticated maneuver mode.

2. OUTER PLANET ORBIT INSERTION PERFORMANCE

Orbit insertion performance characteristics at Saturn and Uranus are presented in Tables F-2 and F-3 for a preliminary multi-mission propulsion size derived from data given in Table F-1 for the Mercury orbit mission. Only results for space-storable propulsion and for a Mariner class payload (680 kg) are listed. For this case the propellant capacity of the propulsion module would be about 700 kg as indicated by the first two rows (under Module B) in Table F-1, see last column.

For the range of trip times covered in Tables F-2 and F-3, 1250 to 1750 days for the Saturn orbiter and 2560 to 4360 days for the Uranus orbiter, the propellant requirements vary over a ratio of more than 2:1 and exceed the available propellant capacity (700 kg) for missions with the shortest trip times in both cases, as indicated by asterisks. Note that in the case of the Saturn orbiter the plane change maneuver requirements are included.

Table F-2. Propellant Required for Orbit Insertion and Plane Change in 1985 Saturn Orbit Mission (Mariner Class Payload; Space-Storable Propellants)

Trip Time Days (Years)	V_∞ (km/sec)	Propellant Requirements in kg (lb_m)						
		Orbit Insertion Maneuver						Plane Change Maneuver
		Variable Thrust Orientation[1]	Fixed Thrust Orientation at Thrust Angle ψ (deg)[2]					
			250	260	270	280	290	
1250 (3.42)	9.73	893*[3] (1969)	974* (2148)	912* (2011)	893* (1969)	912* (2011)	971* (2141)	128 (282)
1400 (3.84)	8.31	656 (1446)	709* (1563)	668 (1473)	656 (1446)	669 (1475)	711* (1568)	142 (313)
1550 (4.25)	7.23	518 (1142)	559 (1233)	528 (1164)	518 (1142)	528 (1164)	561 (1237)	156 (344)
1750 (4.79)	6.22	409 (902)	441 (972)	418 (922)	409 (902)	416 (917)	441 (972)	167 (368)

Assumptions: Saturn orbit dimensions 2.5 X 61.1 R_S
Payload mass 680 kg
Maximum propellant capacity 700 kg } Defined for Mercury orbiter
Propulsion module inert mass 130 kg }
Specific impulse 375 sec
Thrust level

Notes:

[1] Near-optimum thrust orientation, antiparallel to velocity vector

[2] Defined clockwise from radius vector; ψ = 270 degrees antiparallel to velocity at periapsis

[3] Asterisk indicates that propellant mass exceeds propellant capacity of multi-mission module

Table F-3. Propellant Required for Uranus Orbit Insertion (1985 Mission)
(Mariner Class Payload; Space-Storable Propellants)

Trip Time Days (Years)	V_∞ (km/sec)	Propellant Requirements in kg (lb_m)					
		Variable Thrust Orientation[1]	Fixed Thrust Orientation at Thrust Angle ψ (deg)[2]				
			250	260	270	280	290
2560 (7.01)	9.91	829*[3] (1829)	899* (1982)	853* (1882)	837* (1848)	852* (1879)	903* (1991)
2860 (7.83)	8.57	601 (1325)	644 (1421)	616 (1357)	605 (1335)	614 (1354)	643 (1418)
3260 (8.93)	7.23	431 (951)	453 (1008)	438 (965)	433 (955)	439 (967)	456 (1006)
3660 (10.02)	6.25	334 (736)	354 (781)	339 (747)	335 (739)	339 (747)	353 (779)
4360 (11.90)	5.21	251 (554)	269 (592)	255 (563)	252 (556)	256 (564)	267 (590)

Assumed Uranus orbit dimensions 1.1 X 32.1 R_U

Assumptions otherwise identical to those for Saturn Orbiter, Table F-2

Notes:

[1] Near-optimum thrust orientation, antiparallel to velocity vector

[2] Defined clockwise from radius vector; ψ = 270 degrees antiparallel to velocity at periapsis

[3] Asterisk indicates that propellant mass exceeds propellant capacity of multi-mission module

The results show that orbit insertion propellant requirements at both planets are quite insensitive to the selected maneuver mode. Differences between optimum fixed thrust and variable thrust pointing modes are not discernible in the case of the Saturn orbiter, and are 1 percent or less in the case of the Uranus orbiter. Deviations from optimum fixed thrust orientation (tangential to the velocity vector at periapsis) cause only minor performance penalties, i.e., less than 2.5 percent for a 10-degree orientation offset, in both Saturn and Uranus orbit missions.

3. REVISED PROPULSION MODULE SIZING DATA

Results of design iteration and performance analysis of the Mercury orbiter are reflected in the propellant mass, inert mass and tank size data listed in Table F-4. Indicating a size reduction from the values listed previously in Table 4-1 (Volume II), these data conform with the mass values given in Table 7-1.

Table F-4. Propellant Mass, Tank Volume and Dimensions Adopted for Mercury Orbiter

Propulsion Module Type	Propellant Mass* (lb)	Inert Mass* (lb)	Tank Volume* m^3 (in.^{3}) Without Margin	With 15% Margin	Dimensions** cm (in.) 2 Spheres	4 Spheres
Module A						
Earth storable	894 (1971)	209.4 (462)	0.976 (59,478)	1.122 (68,400)	102.1 (40.2)	81.0 (31.9)
Space storable	551 (1215)	175.1 (386)	0.530 (32,312)	0.609 (37,159)	74.0 (32.9)	58.7 (26.1)
Module B						
Earth storable	1272 (2805)	247.2 (545)	1.388 (84,626)	1.596 (97,320)	114.9 (45.2)	91.2 (35.9)
Space storable	781 (1722)	198.1 (437)	0.751 (45,801)	0.864 (52,671)	93.7 (36.9)	74.4 (29.3)

*Each module
**Each tank

Note:
Module A: Fixed thrust angle assumed with 5-degree offset from optimum orientation
Module B: Variable retro-thrust pointing angle assumed

APPENDIX G

DYNAMICS AND ATTITUDE CONTROL OF PROPULSION MODULE A

This appendix considers dynamic and attitude-control characteristics of the selected spinning spacecraft/propulsion module configuration from a feasibility standpoint. Of primary interest are:

- Thrust accelerations
- Deployment and control of the flexible, spin-stabilized spacecraft sun shade in the inbound mission
- The effect of solar pressure unbalance due to addition of the propulsion module and sun shade
- Control of principal axes of inertia in the outbound missions
- Dynamic effects of main thrust application.

1. THRUST ACCELERATIONS

Figure G-1 shows thrust accelerations acting on the flight spacecraft versus spacecraft mass for four thrust levels. Mass variations for the mission classes and propulsion system types for both spinning and nonspinning payload vehicles are indicated at the bottom of the graph. Maximum thrust accelerations are about 0.7 g in the inbound, and 0.16 g in the outbound Pioneer class missions, and 0.48 g and 0.104 g, respectively, for Mariner class missions.

The large acceleration of the Pioneer Mercury orbiter requires retraction of the sun shade to prevent unaccaptable deformations. The payload spacecraft itself (Pioneer Venus) can withstand much larger thrust levels since it is designed for solid rocket thrusts of several thousand pounds in the original Venus orbiter application.

Maximum accelerations occurring in the outer planet missions, by contrast, require some structural stiffening of the payload spacecraft appendages but are readily tolerated by the propulsion module.

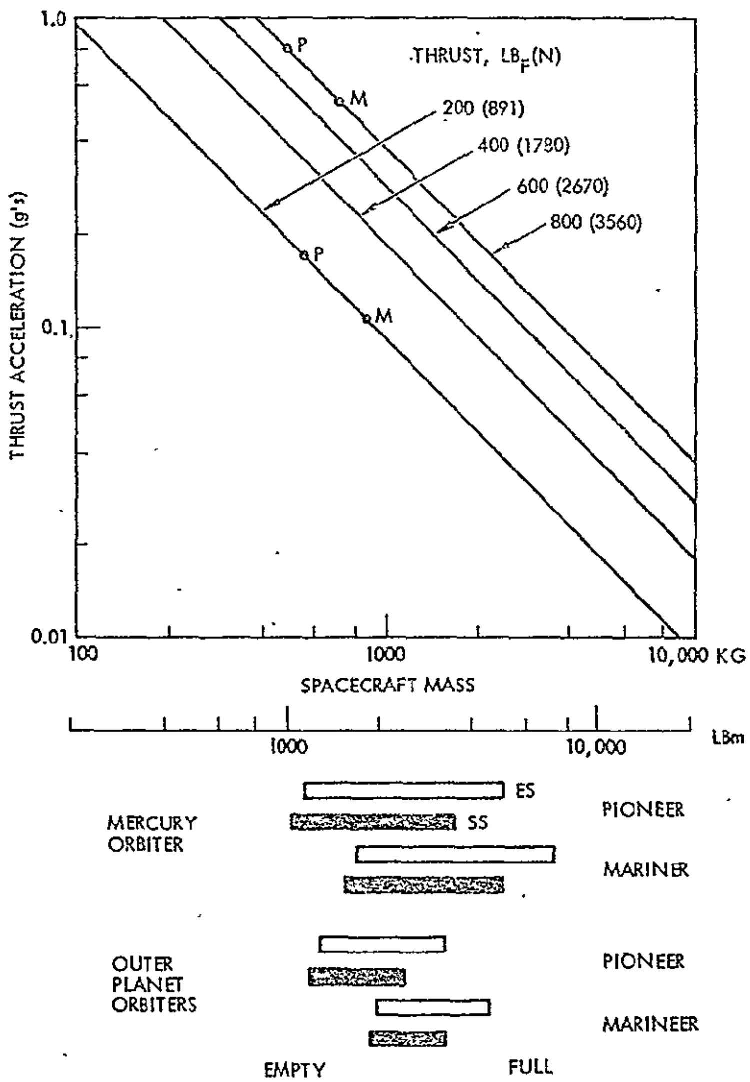

Figure G-1. Thrust Accelerations Versus Spacecraft Mass For Several Thrust Levels

2. SUN-SHADE DEPLOYMENT AND CONTROL (PIONEER MERCURY ORBITER)

Only the Pioneer Mercury orbiter requires a deployed sun shade. The deployment of this flexible structure by centrifugal action is initiated and controlled by individual drive motors, one each per roll-up mandrel.

Slow deployment by the drive motors is necessary to limit deployment transients due to Coriolis effects and to prevent ripping of the shade material when the shade reaches full deployment.

Tension forces in the deployed shade depend on its size and configuration and on the spin rate. The equilibrium between sheet tension, cable tension and centrifugal force in the indented, four-leaf shade configuration shown in the design drawing (Figure 4-12) depends on the angle of attachment of the deployed sheet and, therefore on the depth of indentation. A simplified analysis shows that in first-order approximation the sheet tension is given by

$$F_S = \frac{1}{2} \frac{P_c}{\sin(45 + \alpha)}$$

and the total cable tension by

$$F_c = P_c \frac{\sin \alpha}{\sin(45 + \alpha)}$$

where P_c = resultant centrifugal force in each quadrant of the sheet

α = angle between sheet tension force and circular tangent at cable attachment points as identified in diagram, Figure G-2

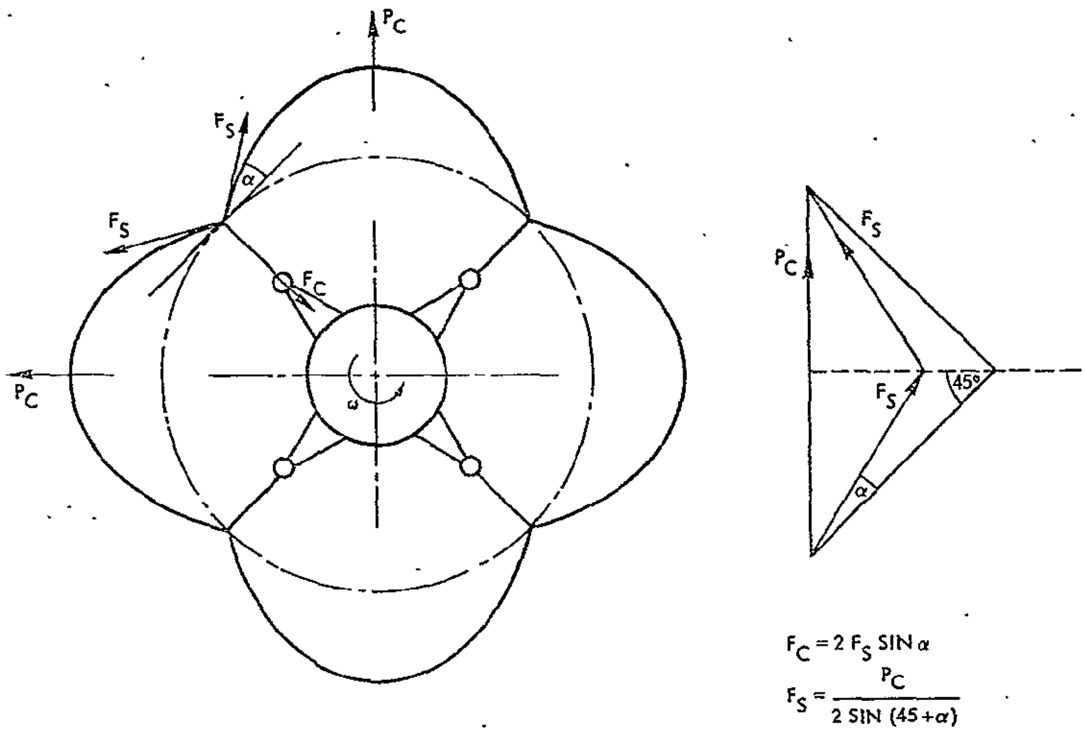

Figure G-2. Force Equilibrium on Deployed Sun Shade

Figure G-3 shows the sheet and cable tensions as functions of the attachment angle α. For zero attachment angle the cylindrical sheet would theoretically be self-supporting with no cable tension acting at the attachment points. Actually, to give stability to the deployed sun shade it is necessary to provide a sizeable cable tension. This produces restoring forces and damping if the sun shade is deflected from the symmetrical steady state configuration as a result of small torques or ΔV maneuvers.

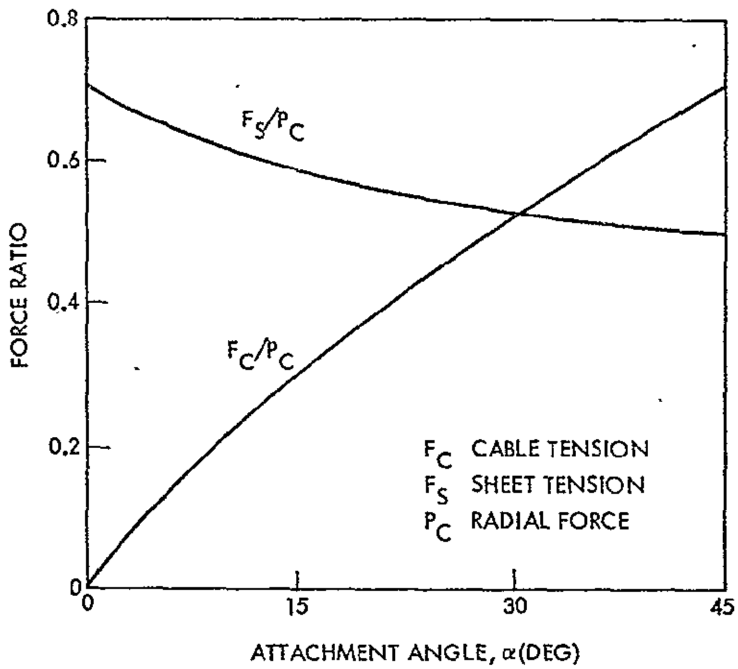

Figure G-3. Variation of Sheet and Cable Tension with Attachment Angle

Figure G-4 illustrates cable deflections due to forces acting parallel to the spacecraft Z axis, e.g., as a result of a precession maneuver by which the sun shade is deflected from its alignment with the X-Y plane. The combined effect of centrifugal forces and cable tensions will restore the sun shade to the steady state position through a series of slow oscillations, dissipating energy through cable and sheet deformations.

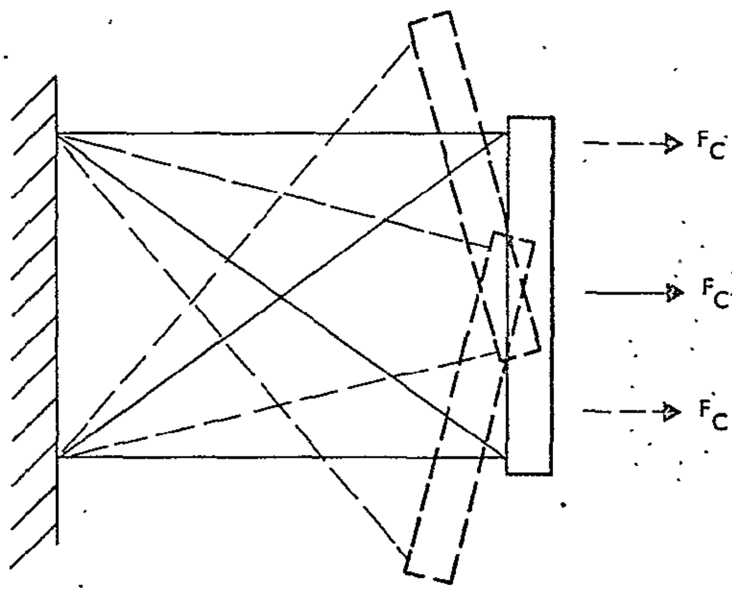

Figure G-4. Deflection of Sun Shade and Retention Cables During Coupled Nutation of Body and Sun Shade

In the design for the space-storable propulsion configuration shown in Figure 4-12 an attachment angle (α) of 28.6 degrees was selected such that the cable tension equals half the centrifugal force per sun-shade quadrant, or 3.5 lb_f (16 N) for the weight, dimensions and nominal spin rate of the system.

An approximate value for frequency of oscillations that would result from a small sun shade deflection, neglecting interaction with the precession of the spacecraft is given by

$$f_S = \frac{1}{2\pi} \sqrt{F_e g / W_S \ell_c} = 0.155 \text{ cps}$$

where W_S = 15 lb_m (6.8 kg) = the mass of the sun shade

ℓ_c = 8 ft (2.44 m) = length of radial cable.

3. PRECESSION MANEUVERS (MERCURY ORBITER)

Actually, a spacecraft precession maneuver leads to coupled oscillations involving the spacecraft and center body and the deployed non-rigid sun shade that are not reflected in the simplified expression given above.

Figure G-5 shows the nature of the dynamic coupling. A precession torque applied to deflect the angular momentum vector $\bar{H}$ by $\Delta\bar{H}_1$ produces a reaction torque from the sun shade retention cables, with the sun shade initially retaining its former inertial orientation. The reaction torque has the effect of introducing a small secondary angular momentum increment $\Delta\bar{H}_2$ oriented normal to $\Delta\bar{H}_1$, which sets up a small nutation. The reaction on the sun shade is to produce a corresponding nutation in opposite direction.

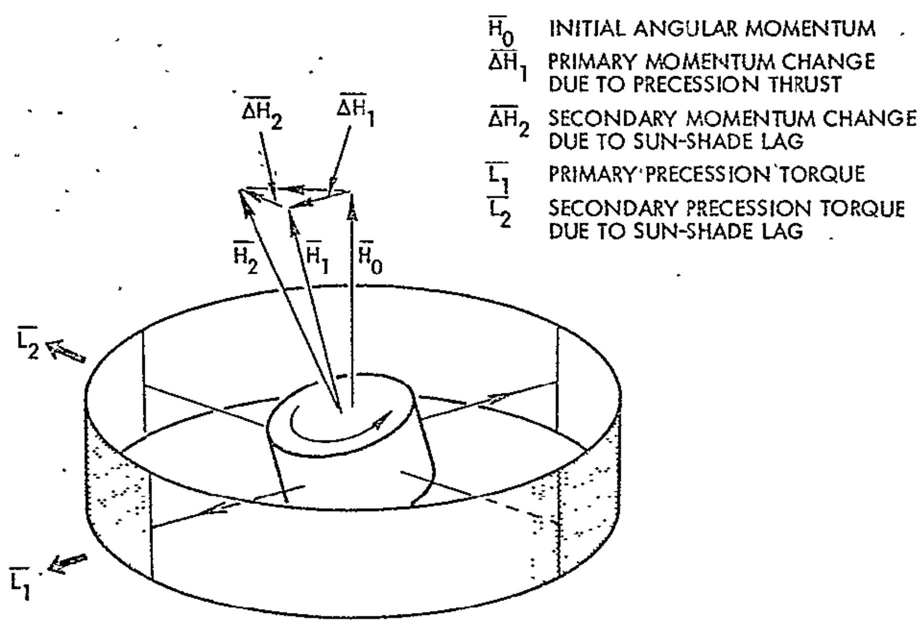

Figure G-5. Effect of Sun Shade on Precession Maneuver

Structural damping and propellant sloshing will ultimately reduce these nutations to zero, with the effect that the sun shade aligns itself with the new spin axis orientation of the center body.

Even without further analysis of the dynamic response of the coupled system, the following qualitative criteria and rules of operation can be deduced:

- Precession maneuvers should be performed infrequently and at a slow rate. (Actually, the nominal cruise orientation can be maintained for long intervals without requiring precession maneuvers.)

- Any precession maneuver is accompanied by slowly damped coupled nutations. Enough time should be allowed for nutations to be damped out before orbit correction maneuvers or attitude-sensitive scientific observations are conducted. The required interval is estimated as about 1 hour.

- The use of teardrop tanks is beneficial in providing increased damping due to propellant sloshing.

- Damping can be further increased by incorporating an appropriately tuned nutation damper, e.g., a mechanism actuated by cable deflections.

As a general rule, other dynamic effects such as angular accelerations during spin-up and despin maneuvers and Coriolis acceleration during shade deployment and retraction sequences can also be minimized by performing these maneuvers at a slow rate. Generally, there are no time constraints demanding rapid maneuver completion.

4. SOLAR PRESSURE UNBALANCE (MERCURY ORBITER)

In the Mercury orbiter mission the large deployed sun shade, with its center of pressure offset by several feet from the spacecraft mass center, causes an appreciable solar pressure unbalance torque. The unbalance torque increases with time as the center of mass shifts upward along the Z axis due to a) propellant depletion and b) first propulsion module staging. Unless counterbalanced by intermittent precession maneuvers, the unbalance torque will cause a spin axis precession in the plane normal to the sun line. Typically, at closest solar distance the precession rate ranges from 50 to 75 degrees per day, depending on whether the sun shade is partially or fully deployed. During the earth-to-Mercury transit phase the unbalance effect and, hence, the precession rate are of course less pronounced.

Unchecked precession of the spin axis is undesirable since it can interfere with effective earth communication. Propellant requirements for intermittent precession maneuvers necessary to retain the nominal cruise attitude are appreciable. Figure G-6 shows the time history of the unbalanced solar pressure torque and the resulting propellant requirements. The figure shows results for three sun shade deployment modes: 1) fully deployed throughout the mission, (2) partially retracted after staging the first propulsion module, and 3) partially retracted and with the lower shade portion jettisoned at the time of propulsion module staging.

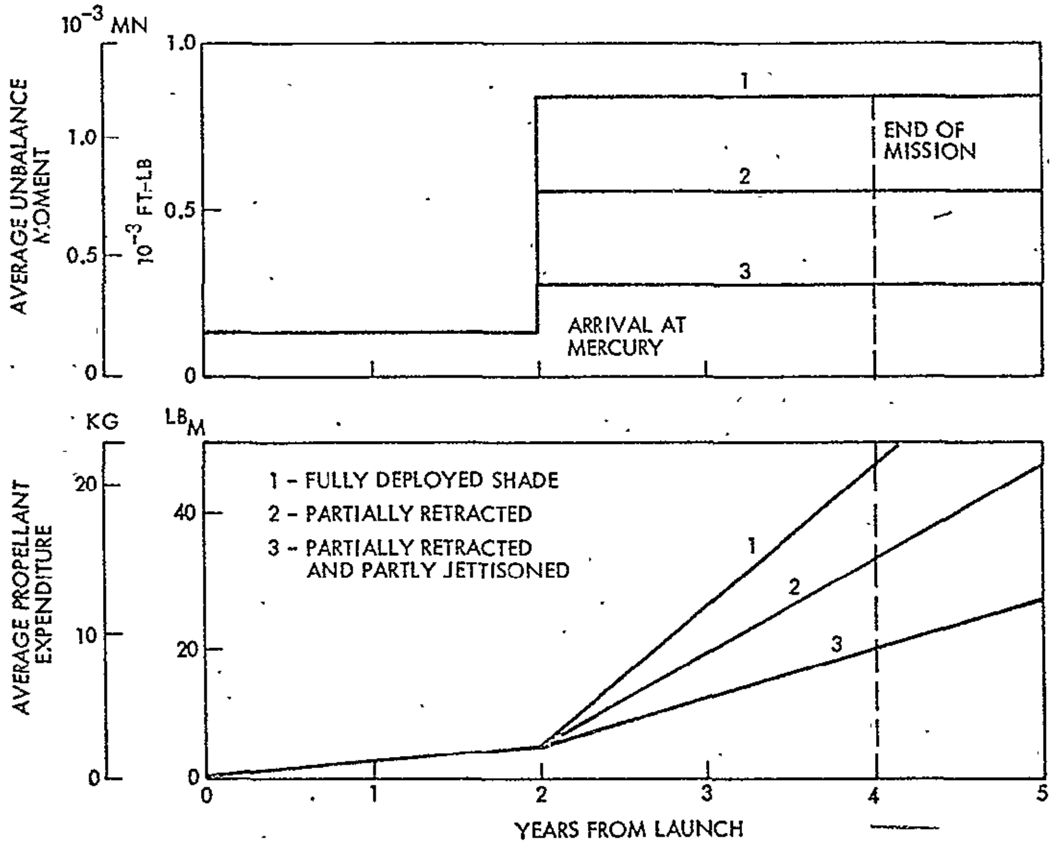

Figure G-6. Average Solar-Pressure Unbalance Moment and Propellant Required for Compensation in Pioneer Mercury Orbit Missions

The expenditure of between 20 and 30 pounds (9.1 to 13.6 kg) of attitude control propellant for unbalance compensation is an unattractive side effect of retaining the second propulsion module and sun shade during the entire orbital mission phase. The option of jettisoning that module when only minor maneuver requirements remain should therefore be seriously considered. This would require that, in the case of the Mercury orbiter, a second set of auxiliary thrusters be carried by the payload spacecraft along with the monopropellant tanks available in its original design.

5. INCREASED SPIN RATE DURING HIGH THRUST MANEUVERS

Spacecraft operation at a higher than nominal spin rate will be required to 1) increase orientation stability during high thrust maneuvers to achieve greater thrust pointing accuracy and reduced residual pointing errors, and 2) to provide additional bending stiffness of deployed appendages against thrust acceleration loads.

Due to unavoidable small thrust vector misalignments, the high thrust maneuver introduces a buildup of precession and nutation angles. After completion of the maneuver, the nutation angle will decay gradually through wobble-damper action and/or inherent damping of deployed structures.

Figure G-7 (A) and (B) show typical pointing errors caused by the main thrust maneuver in Mercury and outer-planet orbiter configurations. The maximum value of the pointing error varies with the inverse square of the spin rate as shown by solid lines. After thrust termination the wobble portion of the pointing error will decay exponentially, leaving a residual pointing error which is shown by the dotted lines in Figure G-7. These results are based on data from the recent Pioneer outer-planet orbiter study (Reference 6). Upper bounds of the pointing error for the Mercury orbiter at the increased spin rate of 30 rpm are 1.2 to 3.5 degrees. For the outer-planet orbiter, at 15 rpm spin rate, they are 0.5 to 0.6 degrees

Spin rate variations due to worst-case thrust misalignment can be as large as ±2 rpm during a large ΔV maneuver with a duration of 25 to 30 minutes. This effect is comparatively small for the selected maneuver phase spin rate of 15 rpm. If a spin rate of only 10 rpm were selected, a 2-rpm deviation would be significant by causing a large (56 percent) increase in maximum pointing errors.

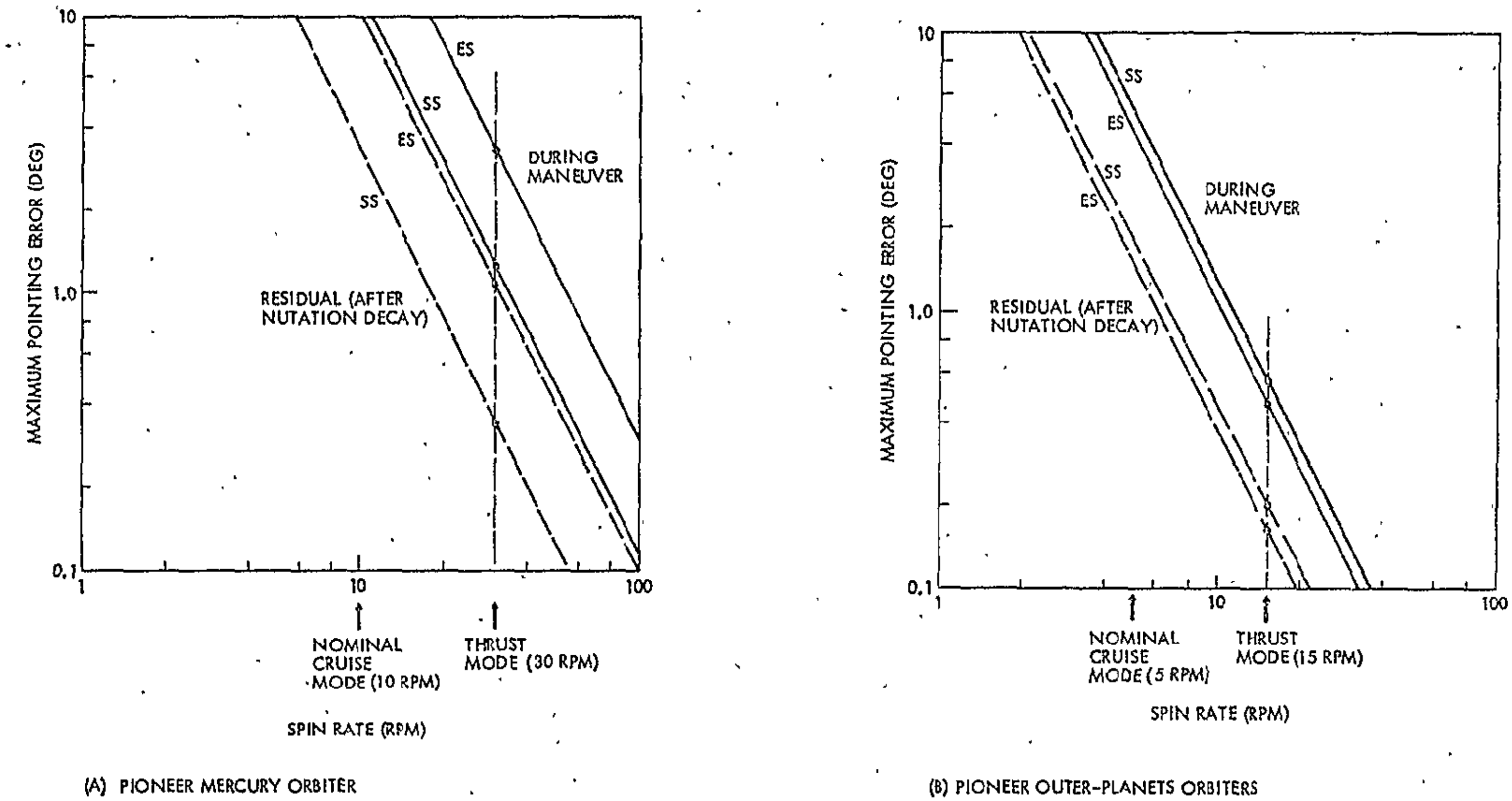

Figure G-7. Maximum Pointing Errors Caused by Main Thrust Maneuvers

6. APPENDAGE DEPLOYMENT OF OUTER-PLANET SPACECRAFT

The Pioneer outer planet flyby spacecraft configuration has an asymmetrical lateral distribution of deployed masses which must be carefully controlled so as to keep the principal axis of inertia oriented parallel to the spacecraft centerline in the deployed configuration. Addition of the large propulsion module lowers the center-of-mass location on the Z axis such that an asymmetrical lateral mass distribution on the payload spacecraft would tend to produce a principal-axis tilt. As a result, unless the principal axis is restored to the centerline, there would be a conical motion of the centerline, degrading high-gain antenna operation.

This can be avoided by assuring that the center of mass of the deployed appendages of the payload spacecraft remains on the centerline in all stages of deployment. This requires that a deployment counterweight be placed at the tip of the 20-foot (6.1-m) magnetometer boom. Secondly, in contrast to the sequential deployment procedure used in Pioneer 10/11, simultaneous deployment is required. The occurrence of large nutation angles during the deployment phase which would impose excessive structural loads on the RTG support arms and the magnetometer boom is thereby precluded.

Results of dynamic analyses performed as part of the Pioneer outer-planet-orbiter study (Reference 6) showed that nutation angles and structural loads can be adequately controlled if start and termination of the deployment phase of the three appendages occur at the same time.

Lateral dynamic loads imposed on the magnetometer boom due to Coriolis acceleration can be adequately controlled by limiting the maximum deployment rate. In consequence, the structural load on appendages due to deployment dynamics can be effectively reduced, and any boom stiffening requirements are largely those due to thrust acceleration.

7. STRUCTURAL STIFFENING OF DEPLOYED APPENDAGES (OUTER-PLANET ORBITERS)

Axial loads on deployed payload appendages induced by high thrust application combine with radial loads due to the centrifugal effect. As the spin rate is increased this leads to an effective stiffening of the deployed appendages against bending due to axial acceleration. Figure G-8 schematically illustrates the stiffening effect due to high spin rates as a result of the vector combination of axial (F_a) and radial (F_r) reaction forces.

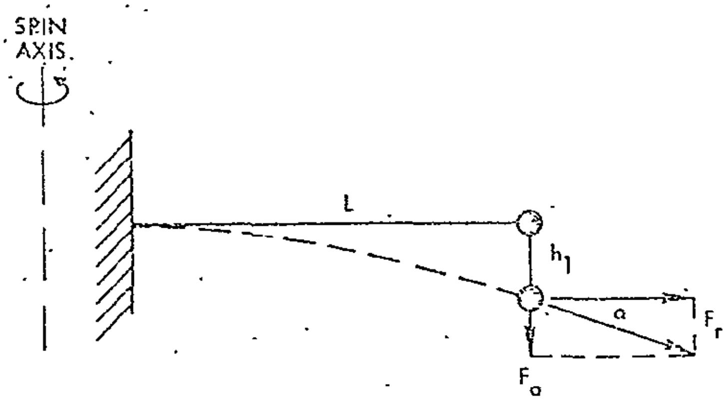

Figure G-8. Cantilevered Boom Under Axial (F_a) and Radial (F_r) Load

The magnetometer boom tends to align itself with the resultant reaction force vector at the tip. Since it is hinged at the root with a ±3-degree deflection range, only boom deflections in excess of ±3 degrees actually induce bending stresses. Previous analysis of bending effects on the appendages of the Pioneer Jupiter orbiter (Reference 24) indicate that the axial and centrifugal load interaction tends to keep the tip deflections of the magnetometer boom and the RTG booms approximately equal. Asymmetry of mass distribution due to boom deflections and, hence, tilting of the principal axis of inertia can thus be minimized.

Consideration was given to the possibility of providing additional stiffening by guy wires extending from deployment reels mounted at the top of the high-gain antenna feed structure. However, this would tend to interfere with wobble damper action by the magnetometer boom, which makes the concept unacceptable.

The present conceptual design relies on structural reinforcement added to the deployment booms and on stiffening due to the increased spin rate.

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